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955 L'ENFANT PLAZA NORTH, S.W. WASHINGTON, D.C. 20024

COVER SHEET FOR TECHNICAL MEMORANDUM

TITLE- An Approach to Manned Space Flight

TM-69-1013-5

DATE- May 13, 1969

FILING CASE NO(S)- 105-3

AUTHOR(S)- D. Macchia
M. H. Skeer

FILING SUBJECT(S)- Common Space Hardware
(ASSIGNED BY AUTHOR(S)-

(NASA-CR-106575) AN APPROACH TO MANNED SPACE FLIGHT (Bellcomm, Inc.) 102 p

N79-71831

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ABSTRACT

An integrated long range approach to manned space flight based on common hardware and minimal scale missions is discussed. A representative set of lunar, earth orbital, and planetary missions are considered to illustrate this approach. The intent is to examine the feasibility of carrying out a spectrum of missions with a common set of basic systems.

Objectives and overall rationale are stated in the initial sections. Discussion of hardware and typical missions follow. A Mars landing mission is considered in relatively greater detail since new operational and design concepts are introduced.

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SUBJECT: An Approach to Manned Space Flight - Case 105-3

DATE: May 13, 1969

FROM: D. Macchia
M. H. Skeer

TM-69-1013-5

TECHNICAL MEMORANDUM

Summary

The attachment discusses a space program based on common hardware and limited mission scale. The proposed approach enables significant objectives to be accomplished in a variety of mission areas but maintains flexibility to direct emphasis to particular missions. Initial goals include performance of earth orbital, lunar landing, and planetary flyby and landing missions. The approach is to let gross mission goals establish fundamental hardware requirements and allow specific mission characteristics to be derived from hardware capabilities.

Advantages and economies of common hardware include low research and development cost, easier initiation of new programs, low mission hardware cost, reliability, and common support facilities. Major new developments would be reduced by half and benefits of volume production would result. However, these benefits can probably be realized only if programs in the three mission areas are concurrent, overlapping, or closely spaced.

It is contended that mission scale should be set by funding considerations rather than by an arbitrary set of objectives. Consequently, minimum missions are proposed--both in crew complement and experiment payload.

A case is presented which attempts to mechanize an integrated manned program with the fewest possible vehicles and serves to illustrate the potential of such an approach. The hardware elements are the crew support mission module, propulsion module I (utilized for major velocity changes), propulsion module II (employed for abort, attitude control, and low velocity maneuvers) and an earth entry module. Mission peculiar hardware is considered only in one mission area, namely Mars landing, since this hardware would have major impact on such a mission.

The crew support module is a basic module unit adaptable to a variety of missions of varying crew size and duration. Different types of modules can be derived from this basic module by addition or deletion of standardized options. Module weight, size and lifetime constraints are established to assure versatility of module usage and launch on any of several launch vehicles. A single module would support a specialized earth orbital mission or a lunar base; two modules, a planetary mission; three modules, a large multidisciplinary station. Common module subsystems (such as structure, power, life support, environmental control) are generally possible since they are unaffected by the range of operating environments. Different missions will of course require modifications to some subsystems (for example, communications).

Commonality in propulsion system design can also be achieved. Sizing of the propulsion module I is governed by lunar and planetary considerations. Unmanned SV fourth stage and cislunar rescue operations accept derived payload capability. The propulsion module II is sized by planetary abort and lunar surface return applications and is suitable for long term low earth orbit station-keeping operations.

Earth orbital space station options and a lunar base utilizing common hardware elements are described. An approach to accomplish principal planetary exploration objectives of 1) Mars surface sample retrieval, and 2) manned Mars landing at minimal overall cost is also cited. A key element of this plan is commonality of an unmanned Mars surface sample return probe and a manned Mars excursion module. Possible Mars landing mission profiles and probe designs capable of implementing such a program are described.

Mission elements employed for minimum Mars landing missions are Venus swingby, elliptical parking and capture orbits, and direct entry of lander probes. Saving attributed to these mission elements is 75 percent of initial weight in earth orbit.

Acknowledgement

This is a summary document. Vehicle and subsystem design studies, and mission analyses, based on the work of many individuals have been used throughout The extensive list of references notes these contributions and provides detailed backup material. The authors acknowledge J. M. Tschirgi for providing many of the central ideas of the overall plan and much of the supporting arguments. C. L. Davis is acknowledged for contributions to all phases of this effort. Arguments defining and supporting the proposed approach (particularly in the first two sections) are based on a number of general references (1 - 9).

D. Macchia

D. Macchia

M. H. Skeer

M. H. Skeer

$1013-\frac{DM}{MHS}$-nma

Attachments
Main Text
References

ATTACHMENT

OBJECTIVES

PROGRAM GOALS

A primary objective of this document is to illustrate that an integrated manned space program, encompassing earth orbital, lunar, and planetary missions can be achieved at relatively low cost. The suggested approach is based on maximum utilization of a set of common hardware and limiting the scale (and/or objectives) of individual missions. It is noted that common hardware per se does not place limits on the scale of individual missions--a scale limit is a result of cost considerations.

PROGRAM GOALS

A LOW COST SPACE PROGRAM BY:

- COMMON HARDWARE
- LIMITED MISSION SCALE

MISSION GOALS

For many years, perhaps a decade or more, manned space flight will be in a preliminary survey phase to define what can and should be done. Currently this encompasses only low earth orbit experience, with planned cis-lunar and lunar ventures. In the forseeable future, this phase will extend to the near planets. Detailed exploration, experimentation, and exploitation will inevitably follow. But in this preliminary phase the knowledge gained from each mission shapes the objectives of subsequent missions. Consequently, ambitious mission planning and extrapolation of tasks beyond the most obvious preliminary goals is premature. Objectives may change radically after early explorations (1). (Consider for example possible discovery of easily tapped water on the lunar surface). It is therefore desirable that preliminary phases of space flight maintain flexibility in the choice of missions or objectives and limit early commitments of resources.

The proposed approach enables significant objectives to be accomplished in a variety of mission areas but maintains flexibility to direct emphasis to particular areas if requirements materialize. Attention is primarily [illegible] to the achievement of multiple mission goals. This [illegible] overly ambitious planning for particular mission [illegible] could detract from overall objectives.

The integrated program has as its initial goals performance of earth orbital, lunar landing and planetary flyby and landing missions. Eventually distant planet missions will be desirable but are presently beyond the scope of a preliminary exploration program.

MISSION GOALS

REQUIRED	FLYBY	ORBITAL	LANDING
• EARTH		X	
• MOON		X	X
• MARS	X	X	X
• VENUS	X	X	
DESIRABLE			
• ASTEROID	X		
• JUPITER	X		
• MERCURY	X		

RATIONALE

- COMMON HARDWARE
- MISSION SCALE

COMMON HARDWARE

Requirements

The only fundamental design constraints on hardware are those imposed by natural laws. Such constraints are a result of the operating environment, mission length, and a minimal hardware capability for transport of men and payload. Other factors such as objectives, scheduling, mission selection, mission scale, and configurations, are in fact quite flexible and can be controlled.*

For the gross set of mission objectives selected, it is possible to accommodate fundamental constraints with common hardware. The central principles for the approach herein are

- let gross mission goals (noted in the previous table) establish the fundamental hardware requirements
- derive the specific mission characteristics from the hardware capabilities.

The basic approach can be undertaken for any set of mission goals by trading off hardware characteristics and specific mission details. The total plan is shaped by interactions of objectives, requirements and timing of individual missions. Sufficient foresight in early program planning is prerequisite for successful formulation of an integrated program utilizing common hardware (2).

---

*In contrast to NASA which can exercise substantial control in these areas, DOD requirements are dictated in great measure by factors beyond their control, such as performance and numerical strength of weapons systems of potential enemies. Lesser performance, or a schedule slippage in weapons systems to accommodate commonality could result in an unacceptable compromise in overall defense posture. Hence analogies between DOD and NASA development approaches are not generally meaningful.

FUNDAMENTAL REQUIREMENTS

- ENVIRONMENTAL

  (METEOROID, RADIATION, THERMAL)

- MISSION LENGTH

  (EXPENDABLES, RELIABILITY)

- TRANSPORTATION

  (PROPULSION, ACS, G&N)

- CREW SUPPORT

FLEXIBLE REQUIREMENTS

- SCHEDULES

- OBJECTIVES

- MISSION SELECTION

- QUANTITY VARIABLES

  (POWER, CREW, PAYLOAD, DATA RATE, ETC.)

- CONFIGURATION

COMMON HARDWARE

Advantages

Significant economies and advantages result from the use of common hardware.

R & D

Studies have shown that savings due to use of previously developed subsystems in new system assemblies are remarkably small (10). This is due to the fact that "integration"--the problems of adapting a given subsystem to work in the induced environment of others-- is a primary cost in any development. Thus, for maximum R & D cost savings it is advantageous to retain as much of the basic vehicle as possible and achieve multiple usage with additional components.

It is noted that development and production costs for a planetary spacecraft have been estimated to be several times higher than corresponding costs for an earth orbit spacecraft (11,12). It is believed that this comparison incorrectly estimates true potential costs. If the planetary spacecraft were initially used in earth orbit its additional costs (over the earth orbit spacecraft) would be significantly less, since earth orbit missions can also serve as development tests for the planetary spacecraft. Conversely, the earth orbit missions area will benefit from the development of a long duration module by reduced logistics requirements. In general, proper mission timing can permit a more gradual, less costly approach to qualify all space hardware for long term missions.

New Program Initiation

A new mission formulated within capabilities of existing of partially modified hardware can avoid a major new start with attendant administrative and funding problems at the inception phase. The lengthy procedure for program initiation and approval is in part due to the time and expense involved in development of new hardware, and obtaining agreement within NASA as to what this hardware should be. Availability of hardware, if it is generally suitable for the proposed mission, can streamline the administrative process, and in so doing further progress of manned space flight. AAP is case in point. If major new hardware were required the present low cost AAP would probably not have been initiated.

Mission Hardware Recurring Cost

Because of the economies inherent in volume manufacture the recurring costs of mission hardware units can be lower than those of non-common hardware. This will be true even if some hardware units are grossly modified, since commonality exists at some level of sub-assembly. These savings are particularly realizable in the very low volume production associated with space flight.

Reliability

It is axiomatic that the reliability of a hardware system improves with operational experience. Atlas and Titan launch vehicles and the Gemini spacecraft illustrate this principle. Thus, a minimum number of new developments used over an extended time period assures a high level of reliability.

Support Facilities

Commonality can result in considerable operational savings since common facilities, and particularly, more complete utilization of personnel can be achieved. An example of the latter advantage is provided by the SIVB stage which employs common contractor facilities and personnel on both the Saturn I and Saturn V vehicles.

Another significant point worth noting is that fewer developments (due to fewer contractors) considerably reduce the fixed costs of contractor testing facilities and sustaining engineering.

COMMON HARDWARE ADVANTAGES

- LOW TOTAL R & D COST
- EASIER INITIATION OF NEW PROGRAMS
- LOW MISSION HARDWARE COST
- RELIABILITY
- COMMON SUPPORT FACILITIES

COMMON HARDWARE

Extent of Commonality

Application of common hardware in different mission areas requires some compromise, and acceptance of the fact that the mission is to be shaped by the hardware--rather than the spacecraft being configured to preconceived mission requirements.

Common hardware does not mean "identical" hardware. Mission peculiar options, minor modifications, and evolutionary development improvements are considered part of the plan, as discussed later. Furthermore, common hardware does not necessarily fix mission size. Requirements for more crew men or propulsion can be met by additional modules.

Common hardware elements are developed separately over a long period of time commensurate with the overall program plan. For example, the mission module for earth orbit use would precede other systems by several years. The lunar and planetary mission modules would evolve from this basic design. Other mission hardware, particularly for planetary use, need not be developed for some time.

The table estimates the extent to which common hardware may reduce the number of new developments and, in turn, overall development cost. The major potential savings are in the costly modules necessary for crew support, namely the mission and earth entry modules. Both these modules lend themselves to common designs since the requirements for crew support are identical.

MAJOR NEW DEVELOPMENTS

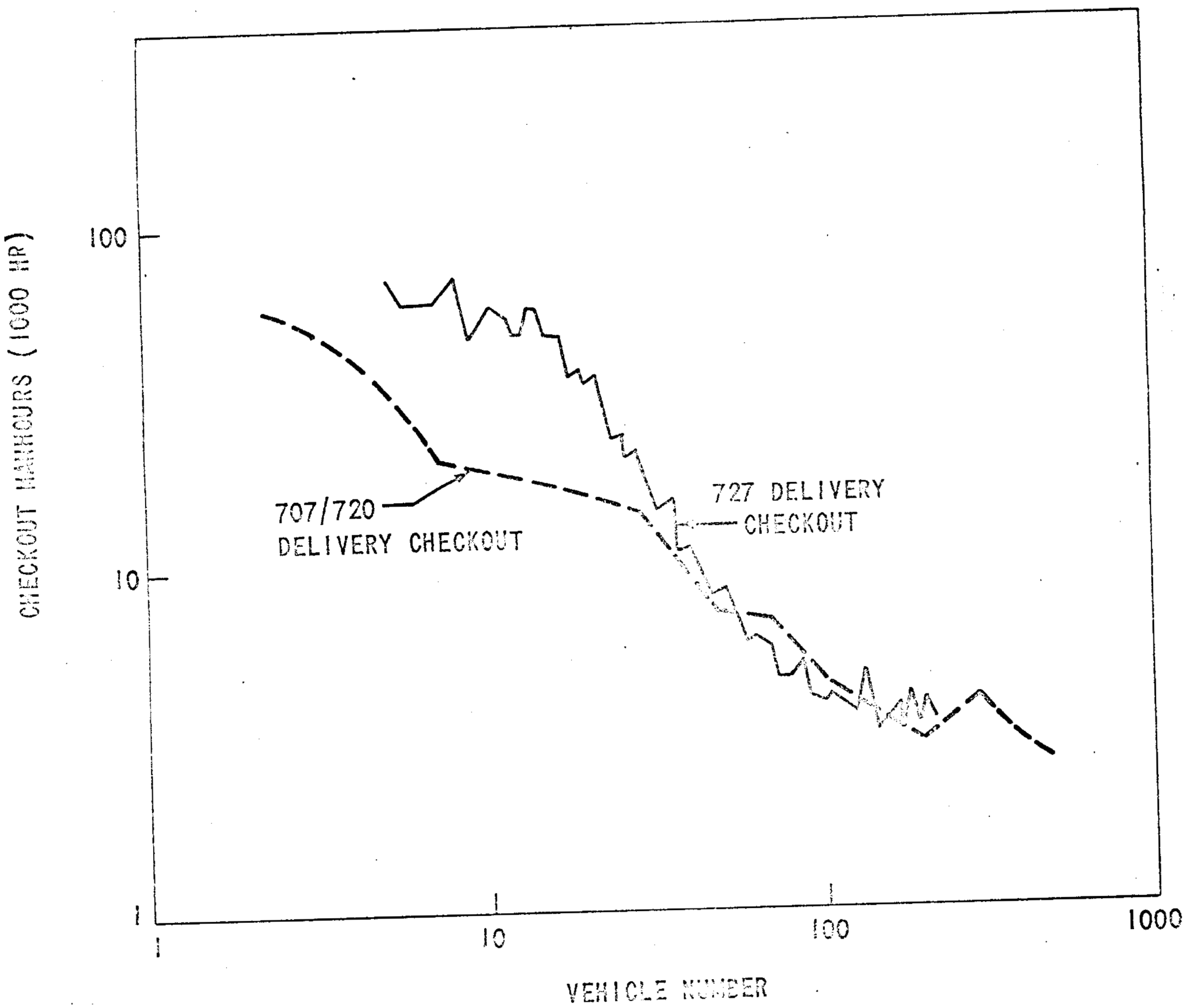
	CURRENT STUDIES	PROPOSED
MISSION MODULE	3-4	1
EARTH ENTRY MODULE	2	1
PROPULSION STAGES	3	2
MARS LANDERS	2	1-2
	10	5-6

COMMENTS

- MISSION MODULES INCLUDE A LUNAR BASE, 1 OR 2 SPACE STATIONS AND PLANETARY MODULES.
- ENTRY MODULES ARE REQUIRED FOR SPACE STATIONS AND PLANETARY RETURN.
- MARS LANDERS INCLUDE UNMANNED SAMPLE RETURN AND A MANNED LANDER.
- PROPULSION STAGES INCLUDE A NUCLEAR OR HIGH ENERGY CRYOGENIC STAGE FOR PLANETARY INJECTION, A LUNAR LANDING STAGE, AND AN ABORT/MIDCOURSE CORRECTION STAGE FOR PLANETARY MISSIONS.

Example: Benefits of Volume Production

An example of decreasing hardware unit costs with vehicle number is given in the chart summarizing Boeing aircraft experience (13). Checkout man-hours, though only a part of total production costs, provides a useful indication of overall trends. These trends are particularly applicable after development problems of the initial prototypes are resolved. The most striking gains are made during the early stages of production. Manned space flight, since it is involved in very limited production, can similarly benefit by increasing its production rates with a limited number of new developments.



AIRCRAFT DELIVERY CHECKOUT EXPERIENCE (BOEING CO.)

COMMON HARDWARE

Example: Evolutionary Development

Use of common hardware does not preclude evolutionary improvements. It has been the experience in many space hardware developments that performance improvements through new technology can be achieved at relatively modest cost. This is, in part, due to the fixed contractor costs (i.e., engineering staff, manufacturing research, etc.) associated with any production run. In practice, these support personnel continually work at product improvement in addition to their regular function of trouble shooting.

The Delta launch vehicle (4) offers an instructive example of evolutionary performance growth while still retaining low cost and reliability. The low orbit altitude payload of the Delta has nearly quadrupled in eight years of operation. This has been achieved for slightly more than $26 million of additional R & D on the initial Delta vehicle. It is noted that a new launch vehicle of 2,300 lb payload would cost perhaps an order of magnitude more to develop.

DELTA LAUNCH VEHICLE COSTS

YEAR	PAYLOAD (LB) TO 200 NMI	ADDITIONAL R & D COST (MIL)	RECURRING UNIT COST (MIL)
1960	600	--	3.09
1962-63	900	6.3	3.09
1964	1300	2.6+*	3.29
1965-66	1600	13.0+*	3.40
1967-68	2300	4.2	4.00
		26.1+	

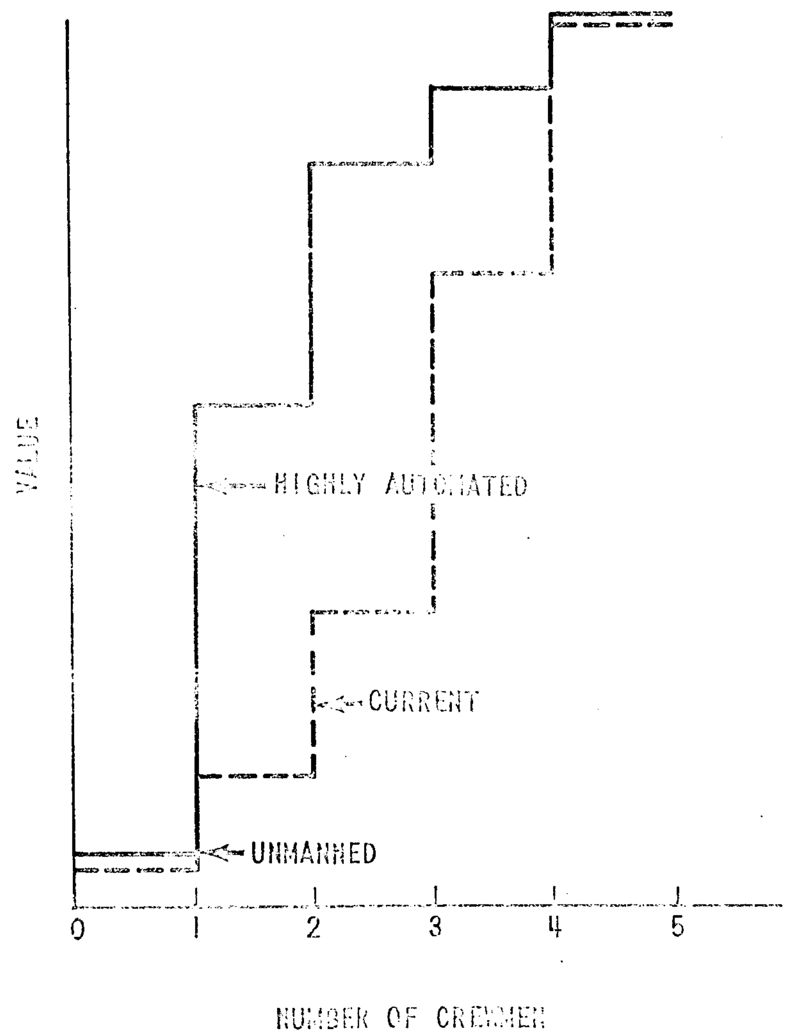
*MINOR AIR FORCE DEVELOPMENT FUNDS HAVE NOT BEEN INCLUDED IN THESE FIGURES.

MISSION SCALE

Value

Questioned here is the necessity for extensive mission scale for early missions--both in crew complement and experiment objectives. Consequently, minimum missions are proposed until such time that more refined objectives can be established. It is believed that the presence of man contributes significantly to any mission by providing real time control (no communication delay), maintenance capability, and reduced automation complexity. A manned mission also permits more complex experiments (large, versatile telescopes or planetary sample return for instance).

The adjoining graph qualitatively illustrates the likely value of mission scale for an early manned mission The solid surve presumes that most of the routine spacecraft monitoring and operation functions are automated. (Apriori there is nothing to preclude this mode of operation if planned on (14).) Thus the first crewman's time can essentially be devoted to experiment or mission objective related tasks. Irrespective of the level of automation, at some point diminishing returns rapdily set in with increasing number of crewmen and experiments. This is because initial objectives can only be more qualitative than quantitative since they are inherently limited by a lack of prior knowledge. The larger mission tends to produce duplication or elaboration in detail.



VALUE
HIGHLY AUTOMATED
CURRENT
UNMANNED
0
1
2
3
4
5
NUMBER OF CREWMEN

MISSION SCALE

Cost

Ambitious mission objectives are synonymous with ambitious funding levels (5). More payload, more experiment man-hours, more data, inevitably means more cost. It is contended that mission scale should be set by funding considerations, rather than by an arbitrary set of objectives.

Compromise of data acquisition and mission modes are equally important considerations. For example, a low circular orbit about Mars or Venus is ideal for purposes of gathering high quality data. But this orbit is extremely expensive in terms of propellant for planetary capture and escape, and thus adds significantly to initial weight in earth orbit. Adoption of worthwhile but less extensive data goals permits use of a highly elliptic orbit at great overall weight savings and thus cost savings.

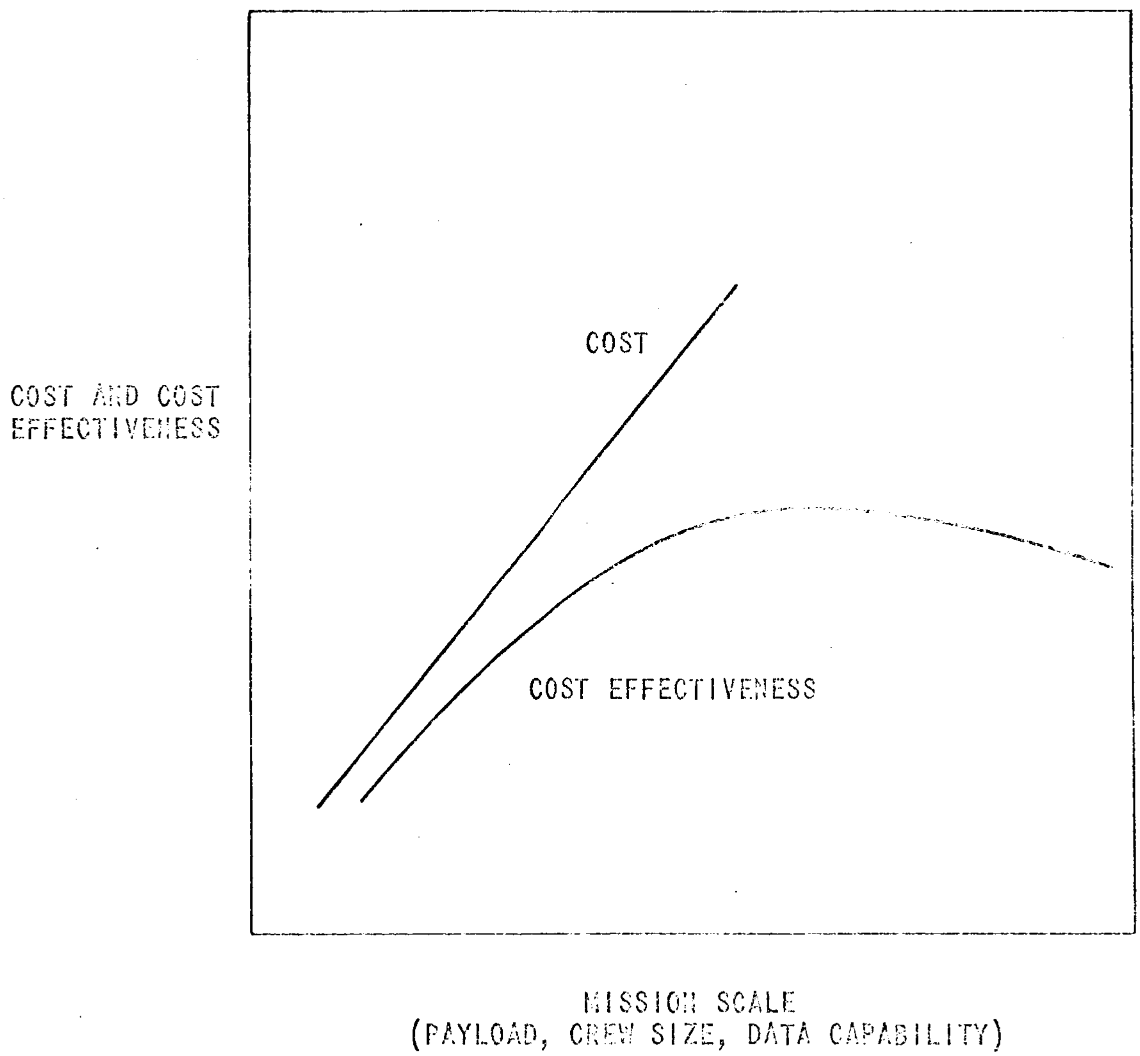
Cost Effectiveness

Ambitious objectives are usually accomplished in a cost effective (i.e., low cost per unit data return or weight transported) fashion with a large system (5). But large systems are costly in an absolute sense.

On any earth orbital research, planetary reconnaissance, or exploratory missions, use of cost effectiveness criteria to establish scale may be questioned since large amounts of data may be unnecessary to the objectives of such a mission.

Furthermore in the context of broader mission goals a cost-effective system for a single mission area is not necessarily desirable. For example, a large space station, unless it is capable of being dissembled into smaller units, is unsuitable for planetary or lunar missions. Thus a smaller unit may be more cost effective on a total program basis.

MISSION SCALE CONSIDERATIONS



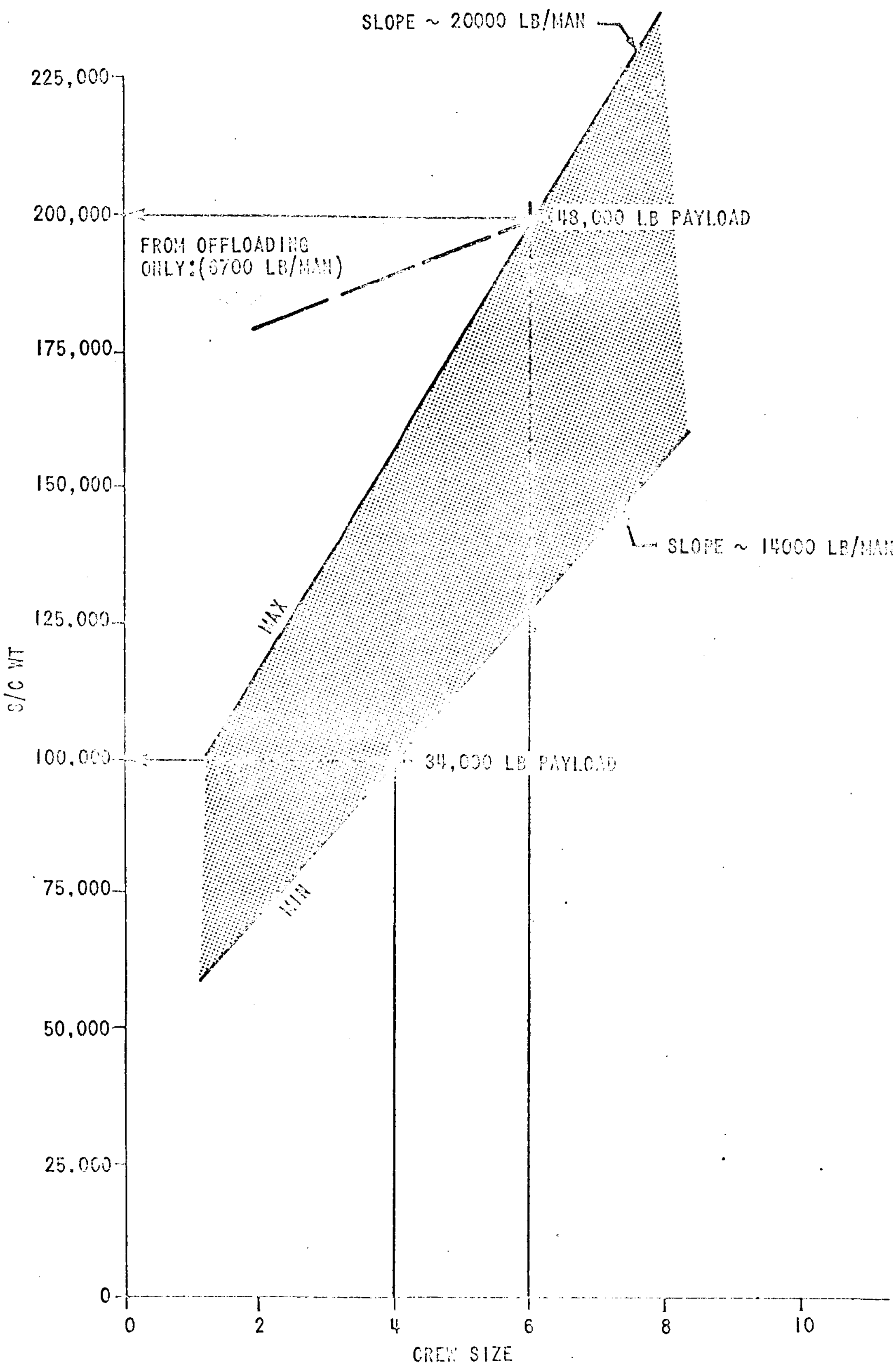
MISSION SCALE

Spacecraft Size

Total spacecraft weight is a useful indicator of cost because of impl... development and recurring launch and hardware costs. The indicated spacecraft weight envelope (15) reflects extensive variation in payload, subsystems, environmental penalties (i.e., meteoroid and radiation protection for different missions), and crew size. Note that weights can diffe by a factor of two or more between two possible planetary spacecraft for a typical mission. Much of these differences are subject to control--particularly the payload and crew size elements.

A minimum weight mission can only be achieved by designing for smaller payload and crew initially. Off-loading a larger spacecraft cannot materially affect potential savings since this only involves expendables and not subsystems.

Weight penalties associated with additional crew-men range from 14,... to 20,000 lbs/man. Such penalties demand a thorough examination of crew size selection criteria. It is evident that m... can be gained by minimizing crew size.



NOTE:

SPACECRAFT WEIGHT IS OPTIMIZED FOR EACH PARTICULAR CREW AND PAYLOAD COMPLEMENT.

COMMON HARDWARE DESCRIPTION

- DESIGN AND DEVELOPMENT PRINCIPLES
- CREW SUPPORT MODULE
  - DESCRIPTION
  - DESIGN CONSTRAINTS
  - APPLICATIONS
- PROPULSION MODULE I
- PROPULSION MODULE II
- EARTH ENTRY MODULE

DESIGN AND DEVELOPMENT PRINCIPLES

Within the spectrum of mission goals, there are many opportunities for commonality--ranging from major subsystems to complete vehicles. The extent to which commonality can be employed is dependent on the flexibility of mission requirements. The case presented attempts to mechanize an integrated manned space program with the fewest possible vehicles and serves to illustrate the potential of this approach (6).

The vehicle configurations presented in this section are by no means the only possible ones satisfying commonality requirements. But they do represent a set of basic vehicles that are capable of performing the different missions.

Analyses were directed to those hardware elements which are capable of multiple mission uses. These elements are the crew support module, earth entry module, propulsion module I (utilized for major velocity changes), and propulsion module II (employed for abort, attitude control, and low velocity maneuvers). Mission peculiar hardware is considered only in one mission area, namely Mars landing, since this hardware would have major impact on such a mission.

VEHICLE USE WITH MISSION AREA

VEHICLES	EARTH ORBIT	LUNAR	PLANETARY	RESCUE
CREW SUPPORT MODULE	• SPACE STATION BUILDING BLOCK FOR ANY ORBIT	• SURFACE SHELTER • ORBIT STATION	• ORBIT AND FLYBY MISSION MODULE	----
EARTH ENTRY MODULE	• CREW LAUNCH & RETURN	• CREW LAUNCH & RETURN • SHORT TERM SHELTER	• CREW LAUNCH & RETURN	CREW LAUNCH & RETURN • MISSION MODULE
PROPULSION MODULE I	• MAJOR ORBIT CHANGES ?	• SHELTER & CARGO LANDING STAGE	• INJECTION STAGE • PLANETARY RETRO & ESCAPE STAGES	• LUNAR LANDING STAGE • SPACE MANEUVERING STAGE
PROPULSION MODULE II	• LONG TERM STATION KEEPING	• LUNAR ASCENT STAGE	• MIDCOURSE Δ V'S ACS, ABORT	• LUNAR ASCENT STAGE • MANEUVERING PROPULSION
MARS LANDER & ASCENT VEHICLE	----	----	• UNMANNED SAMPLE RETURN • MANNED EXCURSION MODULE	----

DESIGN AND DEVELOPMENT PRINCIPLES

The proposed hardware concepts are based on design and development principles which permit versatile long term usage. This versatility is achieved through basic vehicle design, high performance systems, and evolutionary growth.

Basic Vehicle Design

The requirement for weight, size, and environmental compatibility between mission areas is self evident. But less obvious is the initial engineering and manufacturing planning which allows for the incorporation of essential mission peculiar modifications. For example, Propulsion Module I would be designed to accept a lunar landing gear. Similarly, the heat shield on the earth entry module would vary with mission use.

High Performance

Greater hardware capability lessens the imposed constraints on particular missions. An obvious example is the crew support or mission module. High performance is advantageous on any mission because of the reduced propulsion and resupply requirements. In particular, the acceptability of high energy missions (synchronous orbit, lunar base, planetary) to a large degree depends on the ability to perform these missions with minimum propulsion (e.g., a single SV for a synchronous orbit station).

Evolutionary Growth

The missions discussed herein do not require evolutionary uprating in that the initial hardware designs are capable of performing the chosen spectrum of missions. But eventually, a mission requiring greater performance will be identified. The position is taken that the required upratings should preferably be carried out on the basic hardware rather than developing new hardware (i.e., recall Delta launch vehicle experience).

HARDWARE DESIGN AND DEVELOPMENT PRINCIPLES

1. DESIGN FOR MULTIPLE USAGE TO ALLOW
   - MISSION PECULIAR MODIFICATIONS
   - WIDE RANGE OF OPERATING ENVIRONMENTS
   - WEIGHT AND SIZE COMPATIBILITY BETWEEN MISSIONS
2. INCORPORATE HIGH PERFORMANCE TO:
   - REDUCE WEIGHT
   - ELIMINATE RESUPPLY REQUIREMENTS (EXPENDABLES, SUBSYSTEM RELIABILITY)
   - ALLOW NEW MISSION MODES
3. CONSIDER EVOLUTIONARY GROWTH TO:
   - EXTEND USEFUL LIFETIME OF HARDWARE
   - PERFORM NEW MISSIONS

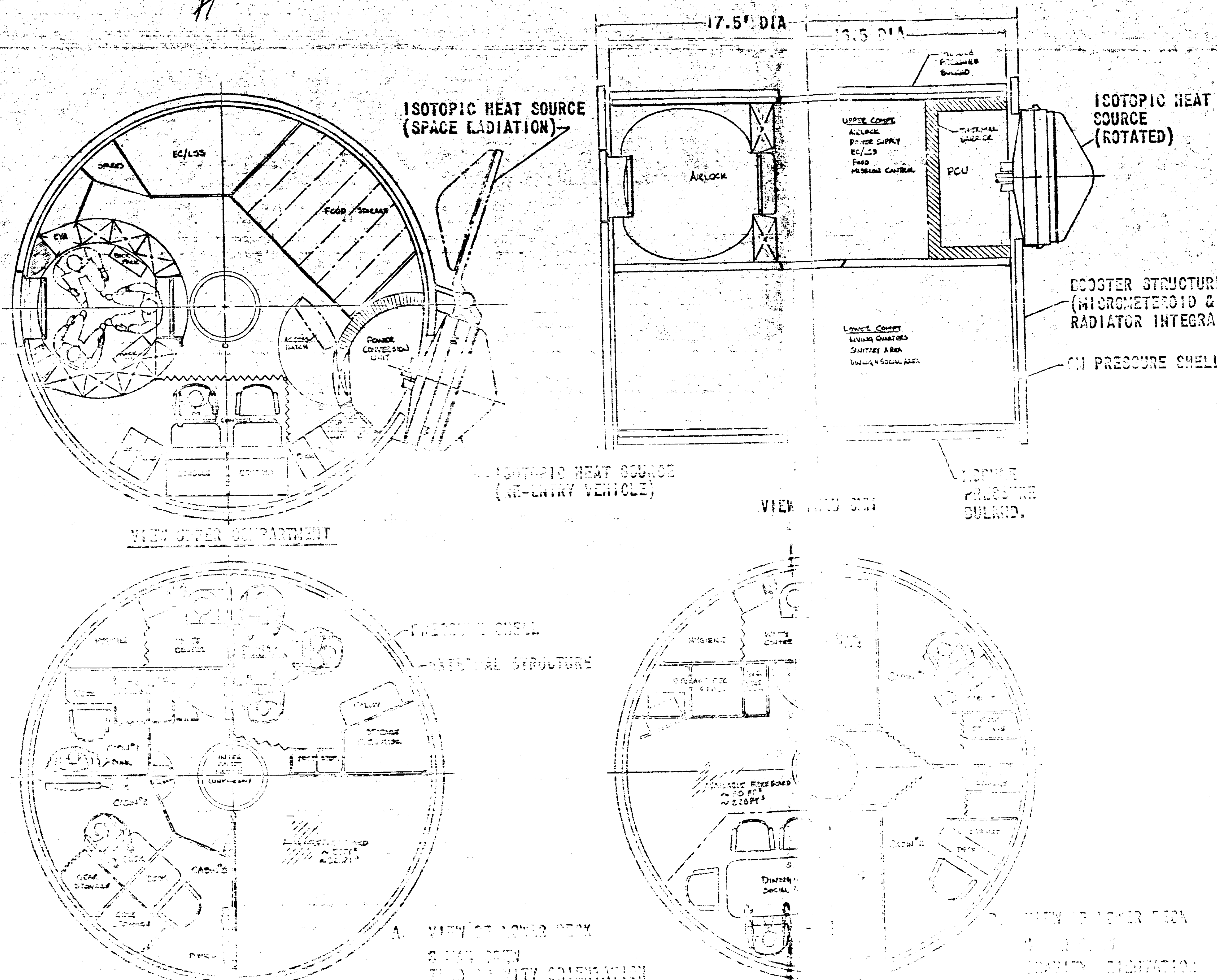
CREW SUPPORT MODULE

Description

The crew support module is a basic module unit adaptable to a variety of missions of varying crew size and duration. Subsystems common to all mission applications include primary structure (pressurized volume), atmospheric systems, thermal control, power, emergency mission control and communications, equipment monitoring and checkout, an airlock, and docking ports. Additional equipment and expendables are incorporated to tailor modules for specialized usage such as mission control, living quarters, or experiment support. Internal architecture (i.e., control center consoles, sleeping quarters, food preparation stations, experiment compartments, etc.) can be mission configured. For example, a module stripped of all living quarters features could provide laboratory space for an assembly of experiments. It is also possible to delete module power units in assemblies of several modules. Propulsion and attitude control functions are supplied by separate propulsion modules.

A representative design of combined mission operations/living quarters module is shown opposite (16). Observe that elimination of living quarters features, control consoles, and the airlock would convert this module into a spacious experiment shell. Alternately, deleting only the control consoles, would free space for additional crewmen. Engineering design and manufacturing planning could permit a number of different module types to be derived from a single basic module design.

FOLDOUT FRAME A



FOLDOUT FRAME B

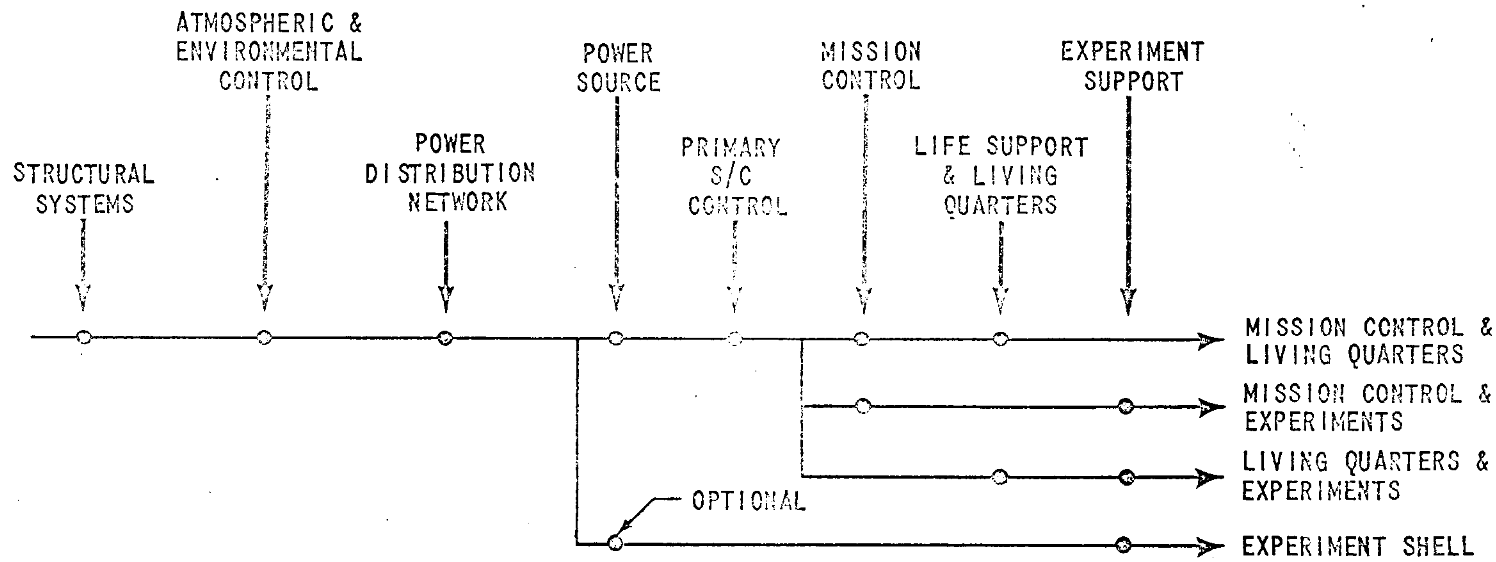
FEATURES

- AN INTERNAL SHELL HOUSING PRESSURIZED VOLUME IS SUPPORTED FROM AN UNPRESSURIZED OUTER SHELL DESIGNED FOR LAUNCH AND DOCKING LOADS (17). THERMAL CONTROL, POWER SYSTEM RADIATORS, AND METEOROID PROTECTION ARE INTEGRAL WITH THE OUTER SHELL (18).
- A COMBINED AIRLOCK - STORM SHELTER COMPARTMENT WITHIN THE MODULE PROVIDES EVA ACCESS, PROTECTION FOR VARIOUS EMERGENCY SITUATIONS, AND ACCESS TO AN EARTH RETURN MODULE. SHIELDING FOR SOLAR STORMS IS ACCOMPLISHED BY PACKAGING FOOD AND OTHER EQUIPMENT AROUND THE AIRLOCK. THE SHELTER IS ADJACENT TO THE OUTER WALL TO INCREASE VOLUME UTILIZATION EFFICIENCY.
- D[illegible] PORTS AT MODULE ENDS AND THE AIRLOCK [illegible] RAPID [illegible]. IN ADDITION TO NORMAL FUNCTIONS [illegible] MODULE DOCKING. ADDITIONAL [illegible] MODULE CIRCUMFERENCE COULD B[illegible] FOR [illegible] OF OUTSIDE VIEWING [illegible].
- [illegible] SHELL WALLS AND SE[illegible] [illegible] ACCESSIBLE FO[illegible] [illegible]
- THE MODULE [illegible] IS CONFIGURED FOR A [illegible] CREW SIZE OF 8. [illegible] A T[illegible] OF [illegible] TIONAL [illegible] COULD BE A[illegible] REQUIRED.
- INDIVIDUAL MODULES HAVE A [illegible] BRAYTON POWER SYSTEM AND [illegible] RETURN PACKAGE (19, 20) [illegible] MODULES PROVIDE ADDED [illegible].

CREW SUPPORT MODULE/DESCRIPTION

Modular Derivatives

Types of module derivatives are established by stages (or paths) of assembly. All modules have identical primary structure (i.e., outer wall, pressure shell, insulation, and support structure); atmospheric and environmental control systems; and a power distribution network. At this point of assembly, the module is an empty shell with only the meteoroid shielding varying to suit the intended mission. Experiments could be housed in this empty shell with data handling services and power supplied by an adjoining module. As options, an experiment module could also include its own power source, primary spacecraft control, and mission control. This optional module assembly can separate mission operations from crew living space if an additional living quarters module is provided on the mission. The living quarters module is derived from an empty shell by adding crew and life support systems (i.e., bunks, galley, hygiene areas, etc.), and a power source.



TYPICAL MODULE ASSEMBLY PATH

CREW SUPPORT MODULE

Design Constraints

The proposed mission uses of the crew support module establish several basic module design constraints which are summarized in the adjoining table. It is noted that in addition to the stated goal of hardware commonality, there are weight, size, and lifetime constraints which assure versatility of module usage.

Common module subsystems such as structure, atmospheric supply and control, power (if isotopic), life support and module control are possible apriori simply because they are unaffected by the range of operating environment. Thermal control, radiation protection, and meteoroid protection subsystems which are environmental dependent can also be common as will be shown. However, the different missions will require some degree of modification to some subsystems (i.e., communications, data handling, guidance and navigation).

Versatility of module usage is essential to the feasibility of the overall program plan. Therefore, maximum module weight limits which allow lunar base or low weight earth orbit missions are necessary. Module dimensions that are comparible with any of the several launch vehicles are also required.

CREW SUPPORT MODULE CHARACTERISTICS

CHARACTERISTIC	SELECTED CONSTRAINTS	DERIVED
SUBSYSTEMS	• COMMON SUBSYSTEMS WHERE FEASIBLE • ADAPTABLE TO MISSION PECULIAR REQUIREMENTS WHERE NECESSARY	• COMMON STRUCTURE, ATMOSPHERIC SUPPLY AND CONTROL, POWER LIFE SUPPORT, AND MODULE CONTROL SUB-SYSTEMS. • SOME MODIFICATIONS REQUIRED FOR COMMU-NICATIONS, DATA HAND-LING, AND GUIDANCE & NAVIGATION SUBSYSTEMS.
OPERATIONAL ENVIRONMENT	• OPERATIONAL IN THE ENVI-RONMENT BETWEEN .5 AND 2.0 A.U. INCLUDING EARTH, MARS, VENUS AND LUNAR ORBITS AND LUNAR SURFACE. • 0 TO 1 G OPERATING CAPA-BILITY (LUNAR OR EARTH SURFACE AND SPIN OPERA-TION).	• COMMON DESIGN FOR THERMAL, RADIATION, AND METEOROID ENVIRON-MENT. • CONFIGURATION WITH "FLOOR & CEILING".
LIFETIME CONSIDERATIONS	• PLANETARY, LUNAR, AND LONG TERM EARTH ORBITAL MISSION DURATION CAPA-BILITY.	• 2 YEAR INDEPENDENT OPERATION CAPABILITY.
WEIGHT	• 3 MAN - 2 YEAR ORBITAL SPACE STATION MODULE CAPABLE OF TIIIM LAUNCH TO LOW POLAR ORBIT. • 3 MAN - 1 YEAR LUNAR BASE COMPATIBLE WITH LANDED LUNAR PAYLOAD OF STANDARD SV (45,000 LB. WITH PMI).	• <32 K GROSS WEIGHT WITH EXPENDABLE OFF-LOADING. • <35 K GROSS WEIGHT WITH REDUCED CON-SUMABLES AND SUB-SYSTEMS TO ALLOW LUNAR CARGO OF 10,000 + LB.
DIMENSIONS	• COMPATIBLE WITH INT-20, INT-21, SIB AND SV LAUNCH VEHICLES. • MAXIMUM MODULE DIMENSIONS FIXED BY TIIIM HAMMERHEAD OR SLA PACKAGING LIMITS (21).	• <21.7 FT DIAMETER • ~17.5 FT DIA. BY ~14 FT LENGTH

CREW SUPPORT MODULE/DESIGN CONSTRAINTS

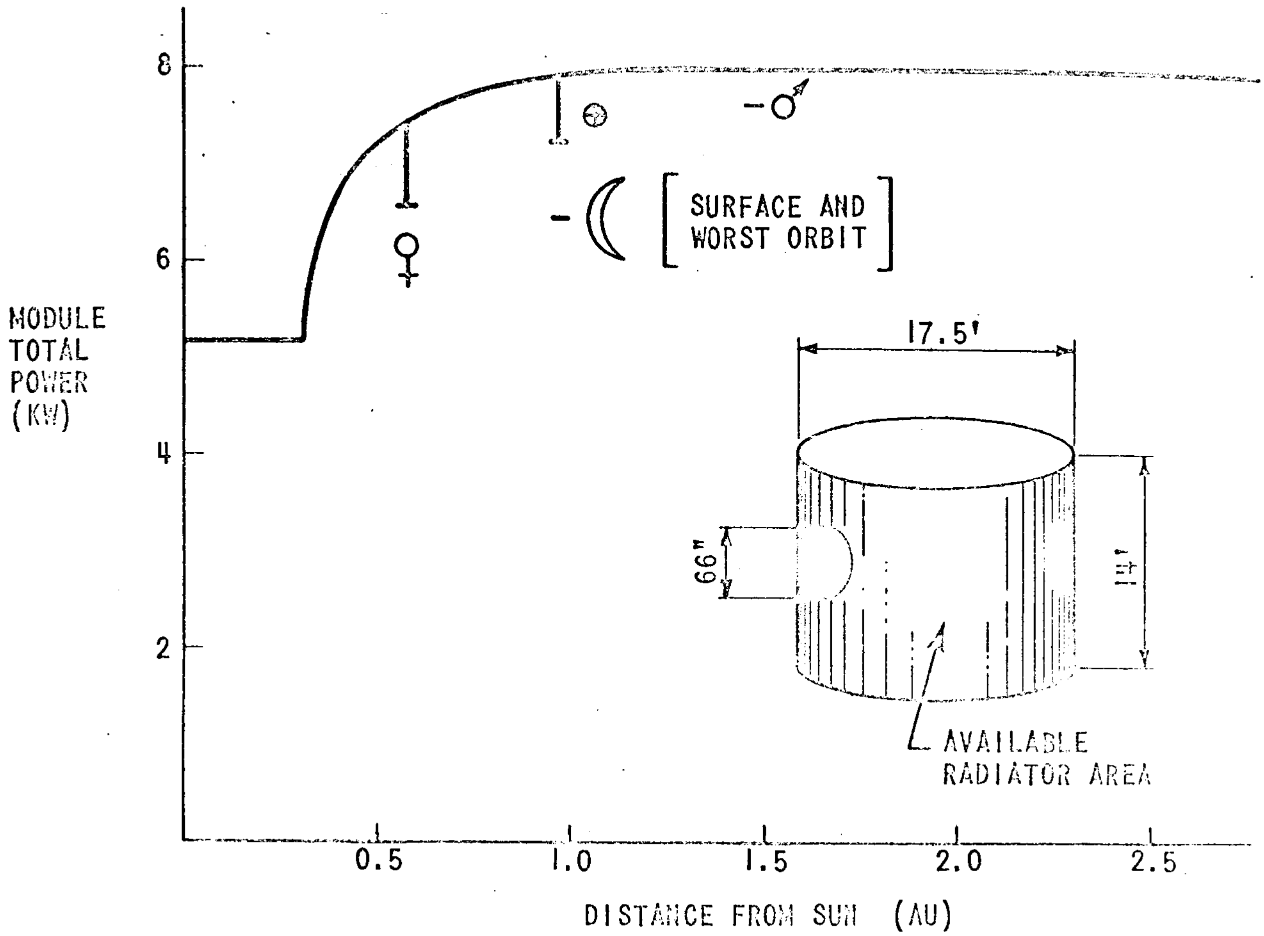
Operational Environment--Thermal Control

A common spacecraft thermal control system can operate throughout the selected mission range (.5 au to 2 au including planetary orbits and the lunar surface) with proper thermal coating and sufficient radiation area to dump power and ECS subsystem heat generated onboard (22). Major advances in thermal coating, namely development of optical solar reflection material ($\alpha = .05$, $\varepsilon = .77$) with excellent long term stability characteristics make the module thermal control system essentially independent of the operating environment.

The module configuration selected can radiate sufficient heat from its outer wall to accommodate the power system heat rejection and ECS heat rejection for a cabin power input of at least 7 kwe. The actual power level depends on power system efficiency, ECS & PCS rejection temperatures, radiator area and properties, and the position in space. The capabilities of the assumed configuration are illustrated.

On this basis a common thermal design can be used for the entire spectrum of potential missions. Minor modifications are required for the lunar surface, low lunar orbit, and possible missions within .5 au of the sun. These modifications are:

- On the lunar surface, a solar reflective mat or an additional radiator must be deployed and OSR material must be used on the CMM surface.
- In a low lunar orbit, a transient technique such as heating and cooling storage water is required to handle hot portions of the orbit.
- For possible missions less than .5 au the spacecraft must be end oriented to the sun.



BASED ON

- ISOTOPE BRAYTON POWER SYSTEM, η = 25%
- HEAT REJECTION TEMPERATURES (T_{ECS} = 530° R T_{PCS} = 650° R)
- OSR COATING (α = 0.05, ϵ = 0.77)
- RADIATOR EFFECTIVENESS = 1.0
- TOTAL RADIATOR AREAS (A_{ECS} = 361 FT^2, A_{PCS} = 361 FT^2)

CREW SUPPORT MODULE/DESIGN CONSTRAINTS

Operational Environmental-Meteoroid Protection

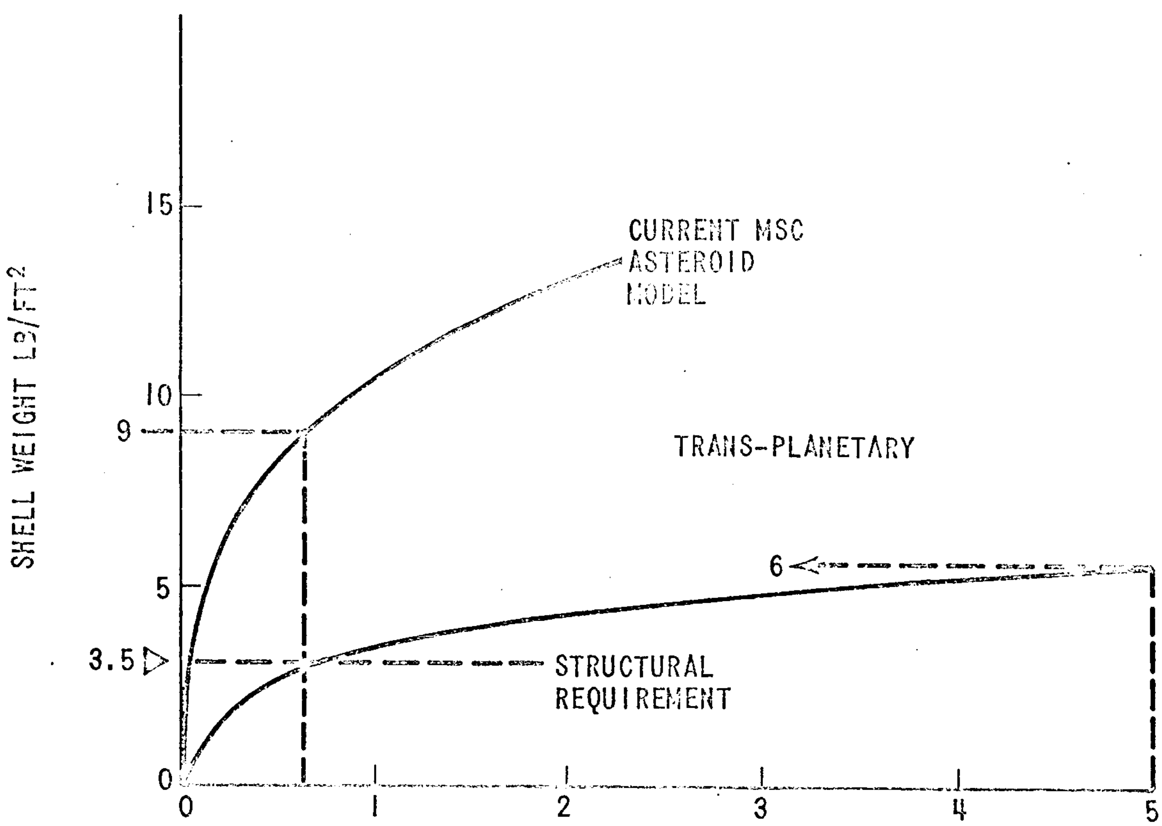
With the exception of the asteroid belts the meteoroid environment in the regions of interplanetary space and near planet can be treated as relatively uniform (23). Hence, for a given configuration, time dependency is the only variable governing meteoroid bumper weight. Wall thickness for missions of less than 1 or 2 years are governed by structural requirements so no additional shielding is required. For missions of extended duration additional bumper protection is achieved by varying outerwall weight (i.e., the thickness of the wall sheets).

The lunar surface environment does not require module modifications. Here, secondary meteoroid impact probability is 3 orders of magnitude greater than that of primary impact; but secondary meteoroid velocities are much lower than primary meteoroid velocities, so that primary meteoroid impact criteria still govern.

Outer wall weight penalties for major planet and asteroid missions through the asteroid belt are significantly increased. Not only is the flux density greater, but mission durations (especially to the major planets) are longer. Such missions are not proposed, but they may be accomodated by a heavier outer wall.

NOTE: TRANS-PLANETARY METEOROID ENVIRONMENT GOVERNS THE OUTER WALL WEIGHT OF THE CMM; THE SECONDARY METEOROID ENVIRONMENT ON THE LOWER SURFACE IS LESS SEVERE

METEOROID ENVIRONMENT



- METEOROID SHIELD GOVERNS OUTER SHELL STRUCTURE
- MISSION DIFFERENCES SMALL
- FOR ASSUMED MODULE CONFIGURATION P (0) = 0.999, BUMPER FACTOR = 5.0

CREW SUPPORT MODULE/DESIGN CONSTRAINTS

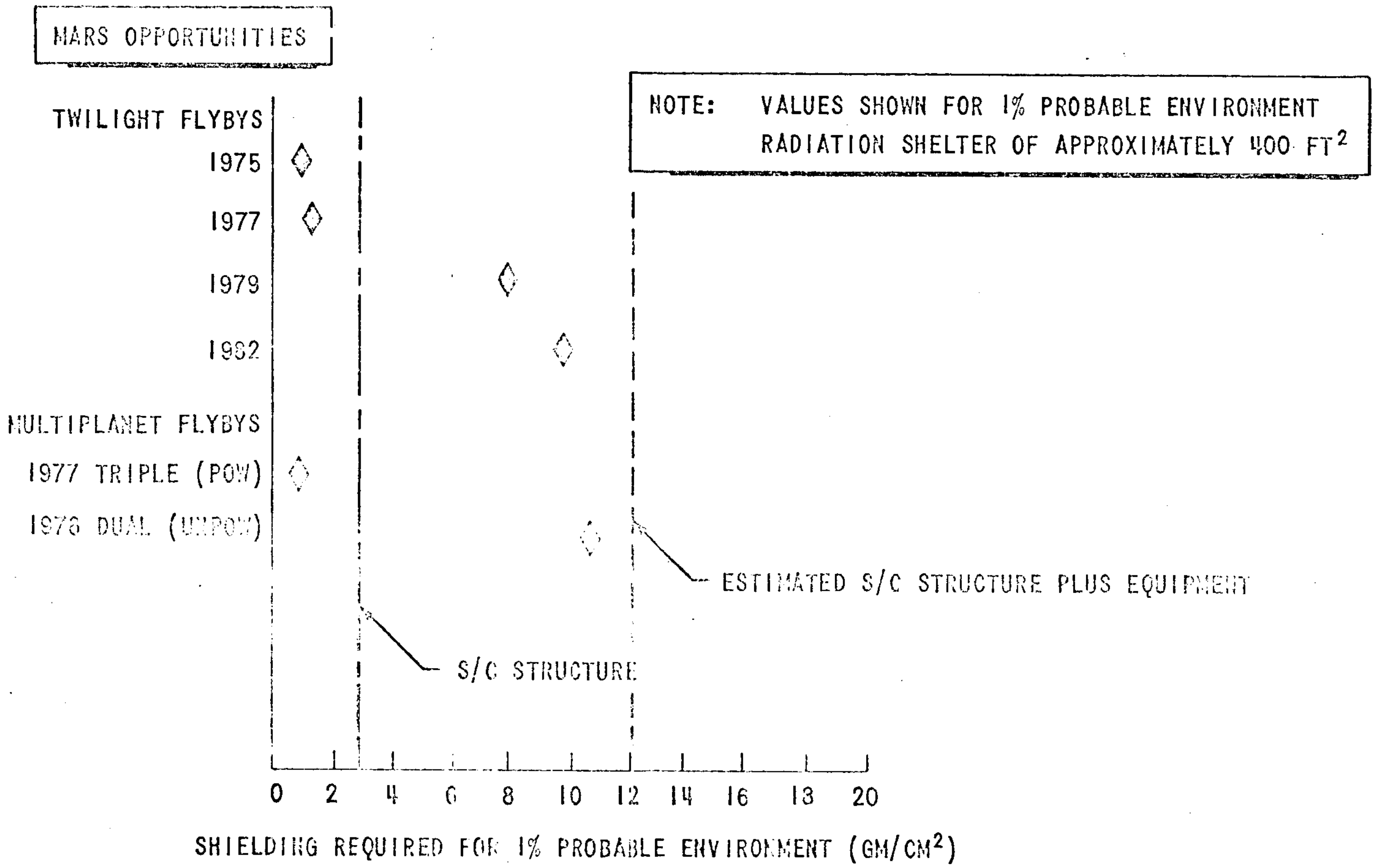
Operational Environment-Radiation Protection

Radiation protection is provided for sporadic solar flare events. Background radiation from the sun and cosmic rays ranging from .25 au and beyond are of sufficiently low energy to be absorbed by the outer spacecraft shell so that the cabin level is maintained within acceptable dose limits without additional protection (24). Storm shelter protection provided by spacecraft structure, equipment storage, and consumables is adequate for baseline mission classes considered. Shielding is maintained by replacing consumables with packaged waste matter.

Specific shielding requirements which are primarily dependent on sun cycle activity are shown for several classes of missions. In all cases design requirements are readily accommodated by storm shelter protection. Additional protection can, if necessary, be provided by added shield weight.

RADIATION PROTECTION

TAKEN FROM NAA/SD
CONTRACT NAS 8-18025



CREW SUPPORT MODULE/APPLICATIONS

The common module defined herein would allow flexibility in mission scale and use (25). A single module, configured for mission control and living quarters, could support a lunar base or a specialized earth orbital mission (i.e., astronomy or earth resources). Two modules (one for living quarters and the other for mission control) could support a planetary mission. Several modules would satisfy the crew size and experiment space requirements of a large multi-disciplinary space station, (e.g., Saturn V launched space station).

Estimated weights based on studies (15, 26, 27, 28) are summarized in the adjacent table.

NUMBER OF MODULES	APPLICATION	CREW SIZE	COMMENTS
1	LUNAR BASE	2-3	1 YEAR BASE
	SMALL SPACE STATION	2-3	• 2 YEAR SYNCHRONOUS, LOW CIRCULAR, OR POLAR • SINGLE DISCIPLINE
2	PLANETARY SPACECRAFT	4-6	• LIVING QUARTER MODULE PLUS CONTROL CENTER & EXPERIMENT CONTROL MODULE
	INTERMEDIATE SPACE STATION	4-6	
3	MULTI-DISCIPLINARY SPACE STATION	6-8	• TWO MODULES PROVIDE CREW QUARTERS AND CONTROL FUNCTIONS. • ONE MODULE HOUSES INTERNAL EXPERIMENTS.

CREW SUPPORT MODULE WEIGHT BREAKDOWN

3 MEN - 2 YRS	MISSION CONTROL & LIVING QUARTERS	LIVING QUARTERS	EXPERIMENT MODULE	EXPERIMENT MODULE
FOOD AND PACKAGING	3800	3800	0	0
ATMOSPHERE SYSTEM	5200****	5200	5200	5200
THERMAL CONTROL	300*	300	300	300
CREW SYSTEM	1500	1500	0	0
WATER AND WASTE SYSTEMS	800	800	0	0
MISSION EQUIPMENT	2000	200**	0	0
POWER (5 kw)	3500	3500	3500	0
SUPPORT STRUCTURE	1700	1500	1500***	1500***
BASIC STRUCT. AND MET. PROTECTION	6000	6000	6000	6000
TOTAL WEIGHT	24800	22800	16500	13000

* 600 FOR LUNAR BASE

** FOR CONTROL OF THIS MODULE

*** FOR EXPERIMENTS

**** GOOD FOR 1 YR - 3 MAN LUNAR BASE WHICH INCLUDES LEAKAGE FROM EVA

PROPULSION MODULE (PM) I

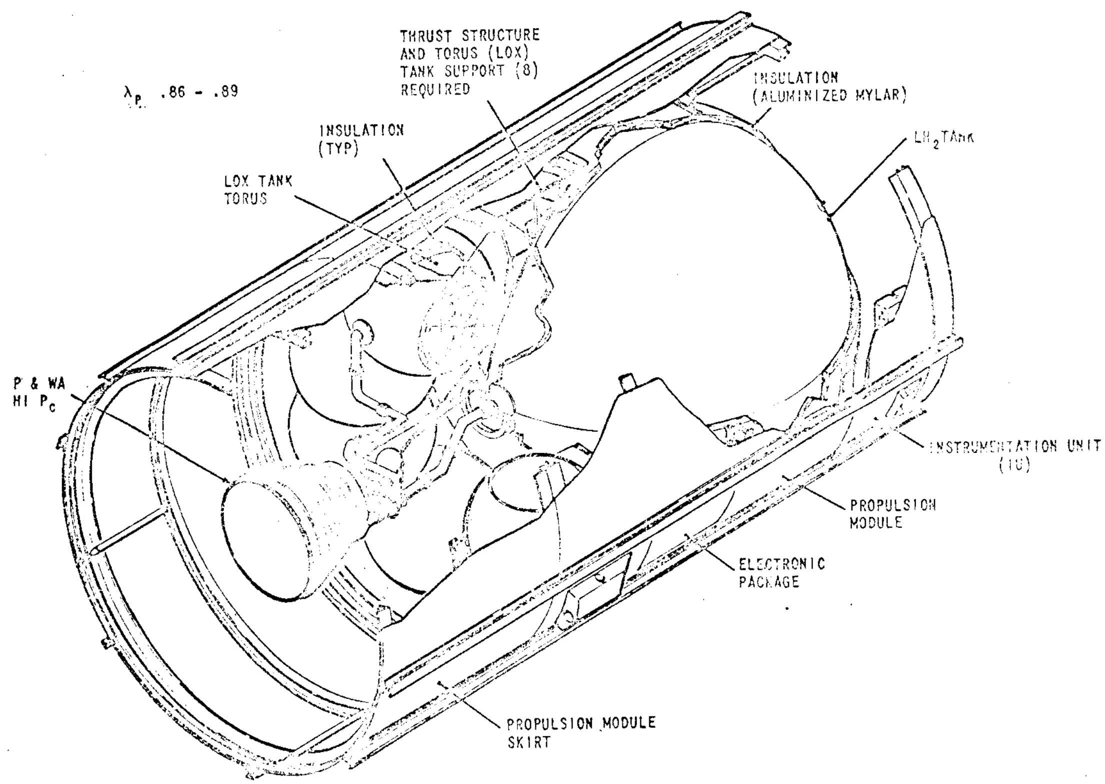
Description and Applications

The PM I (29, 30) is a long term storage LH_2/LO_2 cryogenic stage with a gross weight of 110,000 lbs. Non-vented storage times in excess of 1 year are attainable at 1 au from the sun and in the near earth orbit.

Mission applications for PM I (31) include:

- A fourth stage on a Saturn V capable of landing 45,000 lbs on the lunar surface (exclusive of landing gear penalties).
- A propulsion stage to inject manned interplanetary spacecraft from earth orbit, retro at the planet (Mars or Venus) and escape from the planet for the return trip to earth.
- A fourth stage on Saturn V to launch major unmanned probes.
- A general maneuvering propulsion vehicle in cislunar space for rescue missions.

PM - 1



DOUGLAS STUDY
PM 1 SCHEMATIC

TYPICAL DESIGN

- GROSS WEIGHT ~ 110,000 LB
- MASS FRACTION ~ 0.88
- SPECIFIC IMPULSE ~ 460
- STORAGE TIME (PRIOR TO VENT) ~ 1 YEAR
- LUNAR LANDING GEAR ~ 3000 LB
- THRUST ~ 90 K TO 250 K
- THROTTLE TO (FOR LUNAR LANDING) ~ 10 K

PROPULSION MODULE I

PMI Sizing

Sizing of the PM-I is governed by lunar and planetary considerations. Unmanned SV fourth stage and cislunar rescue applications accept the derived payload capability. Maximum stage performance potential for planetary missions are realized when the stage is used in conjunction with a highly elliptical earth parking orbit (for rendezvous and assembly prior to injection) (32).

Size selection of the PM-I is based on achievement of best overall utilization of launch vehicles for the combined requirements of the mission spectrum cited. Tailored to the needs of planetary missions, a PM-I weight of approximately 220,000 lbs would appear optimum (Figure A) (33). In this case propulsion modules would be launched into a low circular parking orbit. However, a stage of considerably smaller size matched to three stage SV launch capability to highly elliptic parking orbit of 110,000 lbs for standard SV (or 140,000 lbs for uprated SV) could achieve the planetary missions with the same number of SV launches with only modest reductions in planetary payloads (34). Moreover, the smaller stage size is preferable for other applications such as a lunar logistics landing stage (Figure B) and a fourth stage on Saturn V (Figure C), and would also presumably be cheaper to develop.

A PM-I of 110 K matched to elliptical orbit launch capability of the standard SV, is therefore assumed.

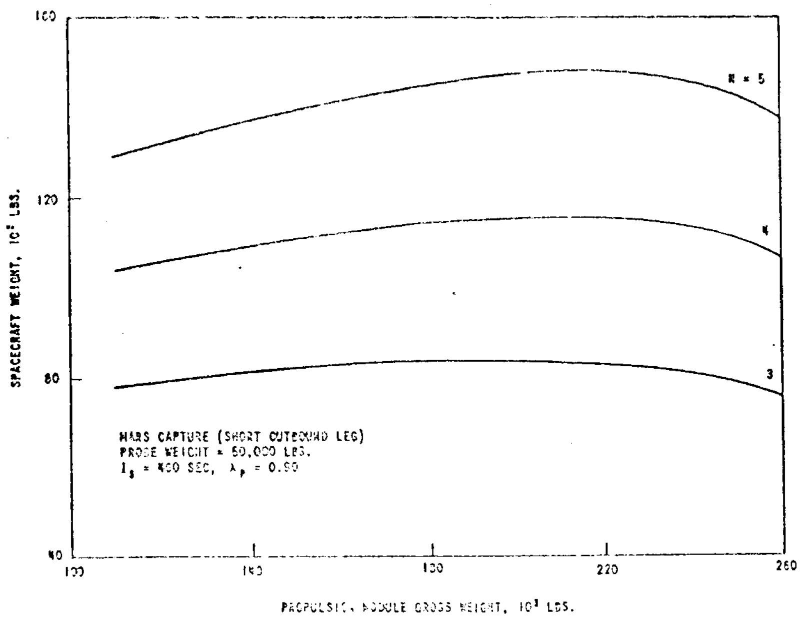


FIGURE A

LUNAR LANDED PAYLOAD VS. STAGE WEIGHT

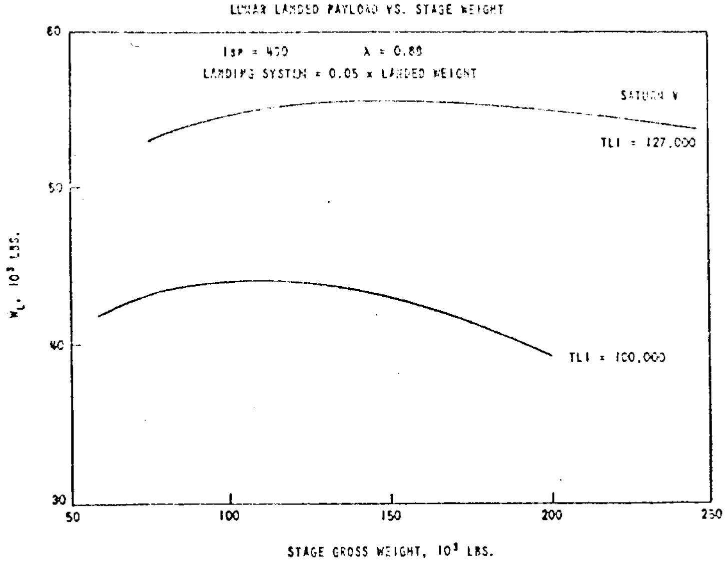
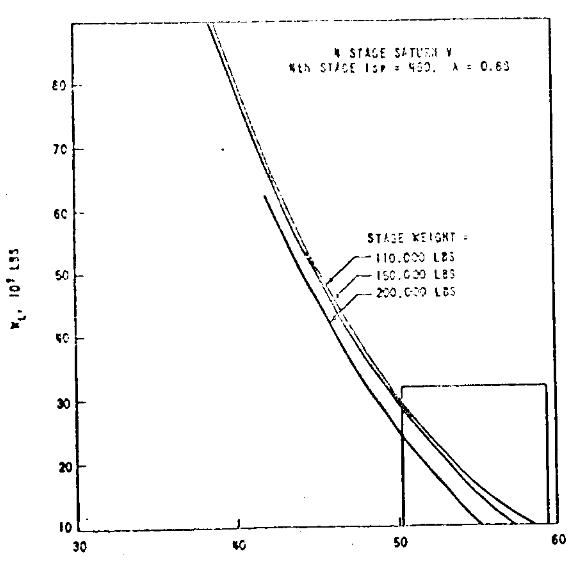


FIGURE B



PAYLOAD VS. VELOCITY

FIGURE C

PROPULSION MODULE (PM) II

Description and Applications

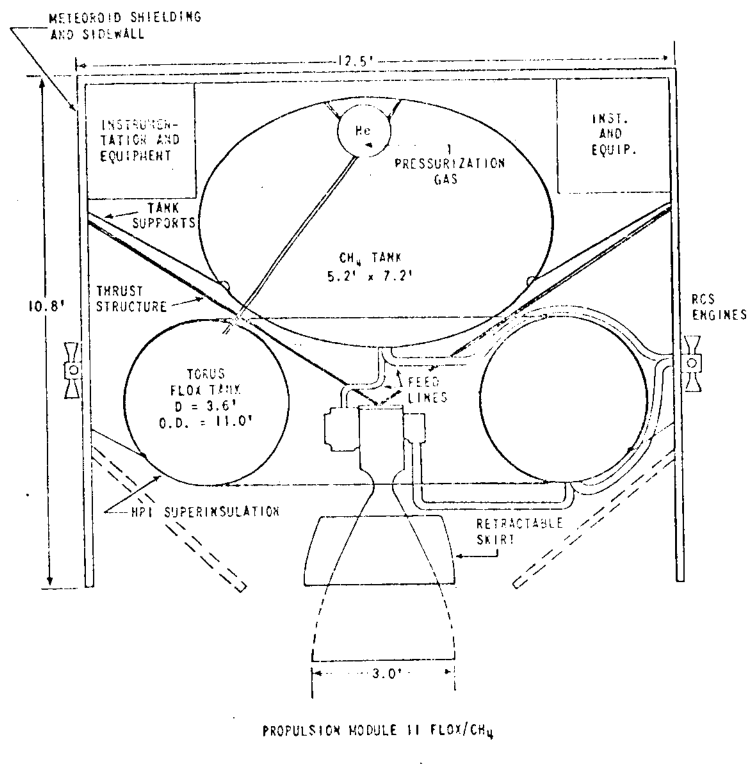
The PM-II is a space storable (FLOX/Methane) propulsion stage of approximately 27,000 lb gross weight (35). It is capable of long term storage (∿ 2 yrs) in space or on the lunar surface. The PM-II performs the following tasks:

- Return, from the lunar surface, of up to 3 men in an earth entry module, plus discretionary payload.
- Abort, from the injection phase of ballistic planetary missions, for up to 4 men in an earth entry module.
- Midcourse corrections and attitude control for planetary spacecraft.
- Station keeping and attitude control for large earth orbit space stations.
- General maneuvering propulsion in cislunar space on rescue missions.

PMII

TYPICAL DESIGN

GROSS WEIGHT	~	27,000 LB
MASS FRACTION	~	.89 TO .81
SPECIFIC IMPULSE	~	400 SEC (FLOX/METHANE)
STORAGE TIME (WITH BOILOFF)	~	2 YRS
THRUST (SET BY ABORT REQUIREMENTS)	~	10-20 K



PROPULSION MODULE II FLOX/CH_4

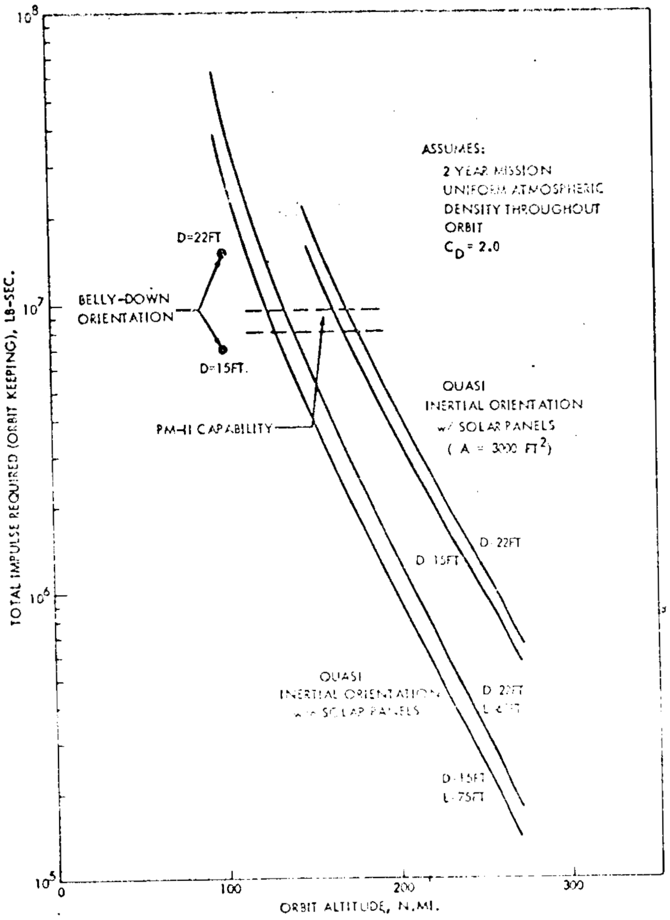
PROPULSION MODULE II

PM-II Sizing

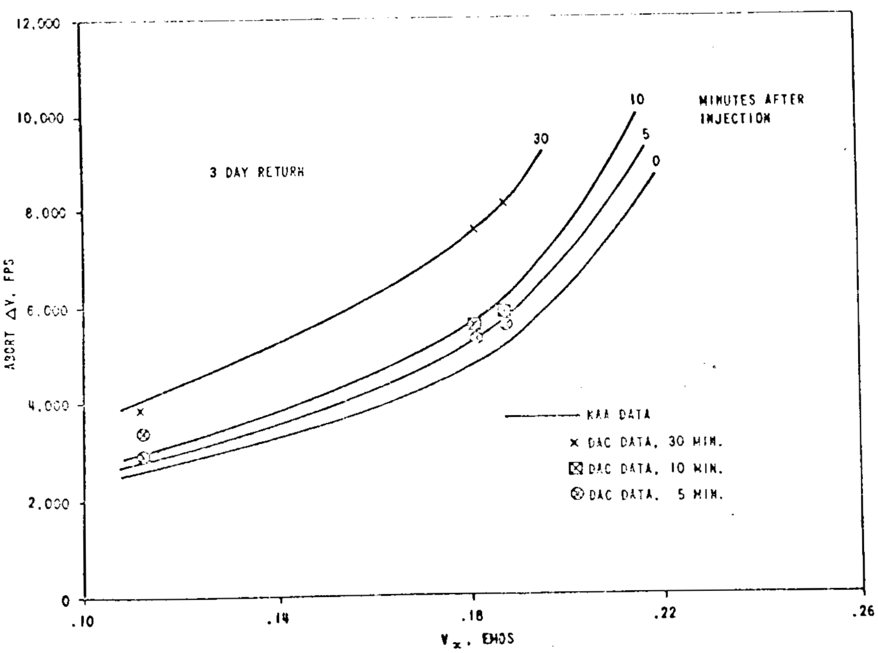
Planetary abort and lunar return have similar propellant requirements for the proposed mission tasks. The abort ΔV selected (Figure A) enables abort from most contemplated planetary mission trajectories (V_∞ < .21 emos), and sizes the PM-II. The derived PM-II provides ample propellant for planetary midcourse corrections.

Little compromise results in PM-II sizing for common lunar and planetary missions usage if crew sizes are similar. However, a larger planetary earth entry module could increase the required PM-II size sufficiently to preclude commonality with the lunar mission. The maximum useable weight of the lunar return PM-II plus earth return module is set by the lunar landed weight capability limit of a PM-I, which is 45,000 lbs.

Earth orbit missions would accept the derived capability of the PM-II (Figure B). It is suitable for two year, low altitude earth viewing missions but grossly oversized for high altitude usage (36). Non-solar array powered space stations can be flown as low as 100 nm (belly down) and inertially fixed stations with large solar arrays could be operated as low as 175 nm for this mission duration.



ORBIT - KEEPING REQUIREMENTS



PLANETARY ABORT REQUIREMENTS

MISSION	PMII - FUNCTION	PROPULSION REQUIREMENTS
LUNAR	DIRECT RETURN 3 MAN EEM (14,200 LB) PLUS PAYLOAD	• ΔV = 10,000 FPS • 1 YR. LUNAR SURFACE STORAGE
EARTH ORBIT	ORBIT KEEPING AND ATTITUDE CONTROL FOR 100-175 N. MI. ALTITUDE SPACE STATIONS	• $I_T \sim 10^7$ LB-SEC (TOTAL IMPULSE) • 2 YR. STORAGE
PLANETARY	MIDCOURSE CORRECTIONS AND ATTITUDE CONTROL FOR 100-200 K S/C	• ΔV ~ 1000-1500 FPS I_{ACS} ~ 500,000 LB-SEC. • 2 YR. STORAGE
PLANETARY ABORT	POST INJECTION ABORT INCL. 4 MAN EEM PLUS LIFE SUPPORT PACKAGE (17,500 + 1700 LBS)	• ΔV < 9010 FPS

EARTH ENTRY MODULE (EEM)

An EEM is capable of multiple usage since the basic function of returning men from space to earth is identical in all mission areas. Structure, life support, and environmental control subsystems would be common. The heat shield, and portions of the communication and G&N systems would be tailored to particular missions. But the basic structure must be designed for the worst case -- return from a planetary mission. Either Apollo EM or low L/D configurations could be utilized.

The EEM would be capable of supporting a crew for up to two weeks, thereby accommodating trans-earth-lunar flights, and aborts shortly after planetary injection. Quiescent storage for up to two years would permit crew return from planetary missions, a long term lunar base, or an earth orbit station. The EEM would be manned during the earth depart phase of all missions for abort during launch or immediately after planetary injection.

Achievement of common EEM usage requires compromise in nominal crew size design selection. A four man (or less) size is proposed, consistent with a policy of reducing mission cost through minimizing hardware weight and crew size. Requirements for larger crews (i.e., earth orbital missions) would be met with multiple modules.

EARTH ENTRY MODULE
DESCRIPTION

- UP TO 55,000 FPS ENTRY VELOCITY CAPABILITY
- DESIGNED FOR 4 MEN
- TWO YEARS SPACE STORABILITY
- UP TO 2 WEEKS ACTIVE LIFE
- LOW L/D OR APOLLO SHAPE

TYPICAL MISSIONS

- EARTH ORBITAL SPACE STATIONS
- LUNAR BASE
- PLANETARY MISSIONS
  - MISSION ELEMENTS
  - EXAMPLE: MINIMUM SCALE MANNED MARS LANDING MISSION
  - POTENTIAL MARS LANDING MISSION WEIGHT REDUCTIONS

EARTH ORBITAL SPACE STATIONS

The relative merits of large multi-discipline versus small single-discipline space stations have not been clearly established (37); nor is there a strong likelihood that they will be established with surety warranting a firm commitment to either one or the other. Small modules which can be clustered into a single large multi-discipline space station preserve the option of assembling a small single module station tailored to specialized tasks.

Current studies of earth orbital missions suggest that only astronomy and medical/behaviroal studies can supply a strong rationale for a long, continuous mission time. Earth resources, physical science and advanced technology appear to be limited programs (i.e., sensor and/or equipment testing, and an "X" man hour experiment program) which may not become continuous manned activities. (Meteorology, an unmanned program, provides an instructive analogy). Small space stations, could for example be employed in early programs, for these limited tasks avoiding a large commitment until need is established. Several inherent advantages include flexible funding (low dollar commitment per mission) and easy experiment integration. If a large station is desired this can be accomplished with modules with reliability proven by continued usage in numerous other missions.

EARTH ORBITAL PROGRAM CONSIDERATIONS

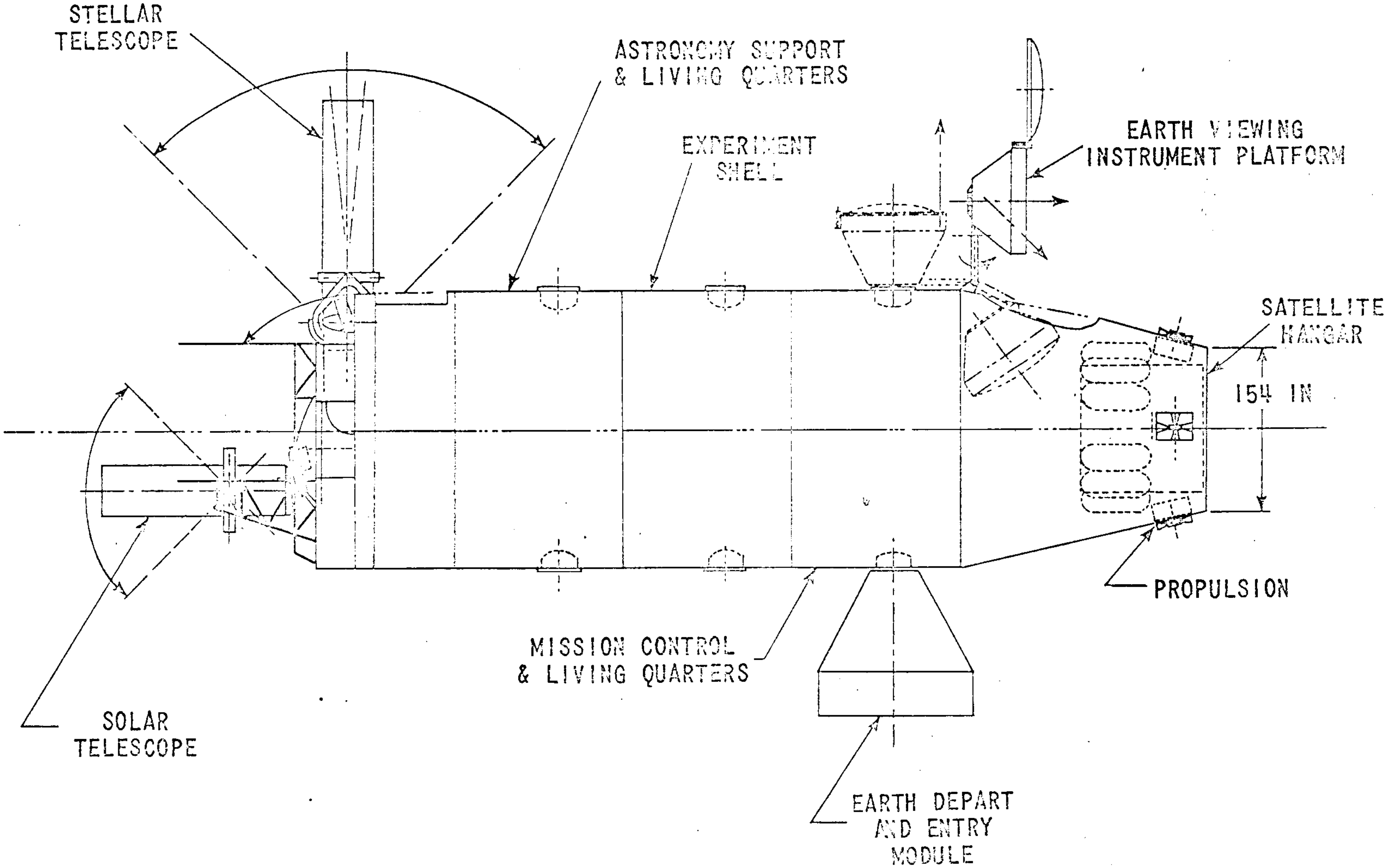
- PROBABILITY OF MISSION SUCCESS WITH SINGLE LAUNCH?
- LARGE INITIAL $ COMMITMENT WITH SINGLE LAUNCH
- SEVERAL SMALL SPACE STATIONS CAN ACCOMMODATE CHANGING PROGRAM FUNDING RATE
- EASIER TO DEVELOP RELIABILITY IN SMALL S/C SINCE MORE ARE LAUNCHED
- SMALL S/C USEFUL FOR OTHER MISSIONS (POLAR, EQUATORIAL OR SYNCHRONOUS ORBITS)
- EXPERIMENT INTEGRATION PROBLEMS OF A LARGE MULTI-DISCIPLINE STATION ARE MORE DIFFICULT AND REQUIRE EARLY DECISIONS ON EXPERIMENTS

EARTH ORBITAL SPACE STATION

Multi-disciplinary Station

The adjacent figure depicts a representative multi-disciplinary space station assembled from three common modules (38). External experiments such as an astronomy facility (39) and an earth sensor package located at opposite station ends are completely accessible from the module area. An unmanned satellite service hangar is incorporated in the earth sensor station section. External experiments are contained within launch vehicle interstage structures. The common modules at the station ends provide living quarters, primary spacecraft control, power, and external experiments. Data handling and communications equipment could be in either end module or duplicated. The center module is an experiment shell which taps power from the end modules. Module docking ports allow use of multiple logistics spacecraft and eliminate the need for a separate docking adapter.

The modular space station has several safety features. Primary spacecraft functions are duplicated in the end modules in addition to living quarters and power. In this way major module failures are isolated from other modules. A mission would be degraded (i.e., return of some crewmen, less power, shorter mission time, loss of some experiment control, etc.) but not aborted.


STELLAR TELESCOPE
ASTRONOMY SUPPORT & LIVING QUARTERS
EXPERIMENT SHELL
EARTH VIEWING INSTRUMENT PLATFORM
SATELLITE HANGAR
154 IN
PROPULSION
MISSION CONTROL & LIVING QUARTERS
SOLAR TELESCOPE
EARTH DEPART AND ENTRY MODULE

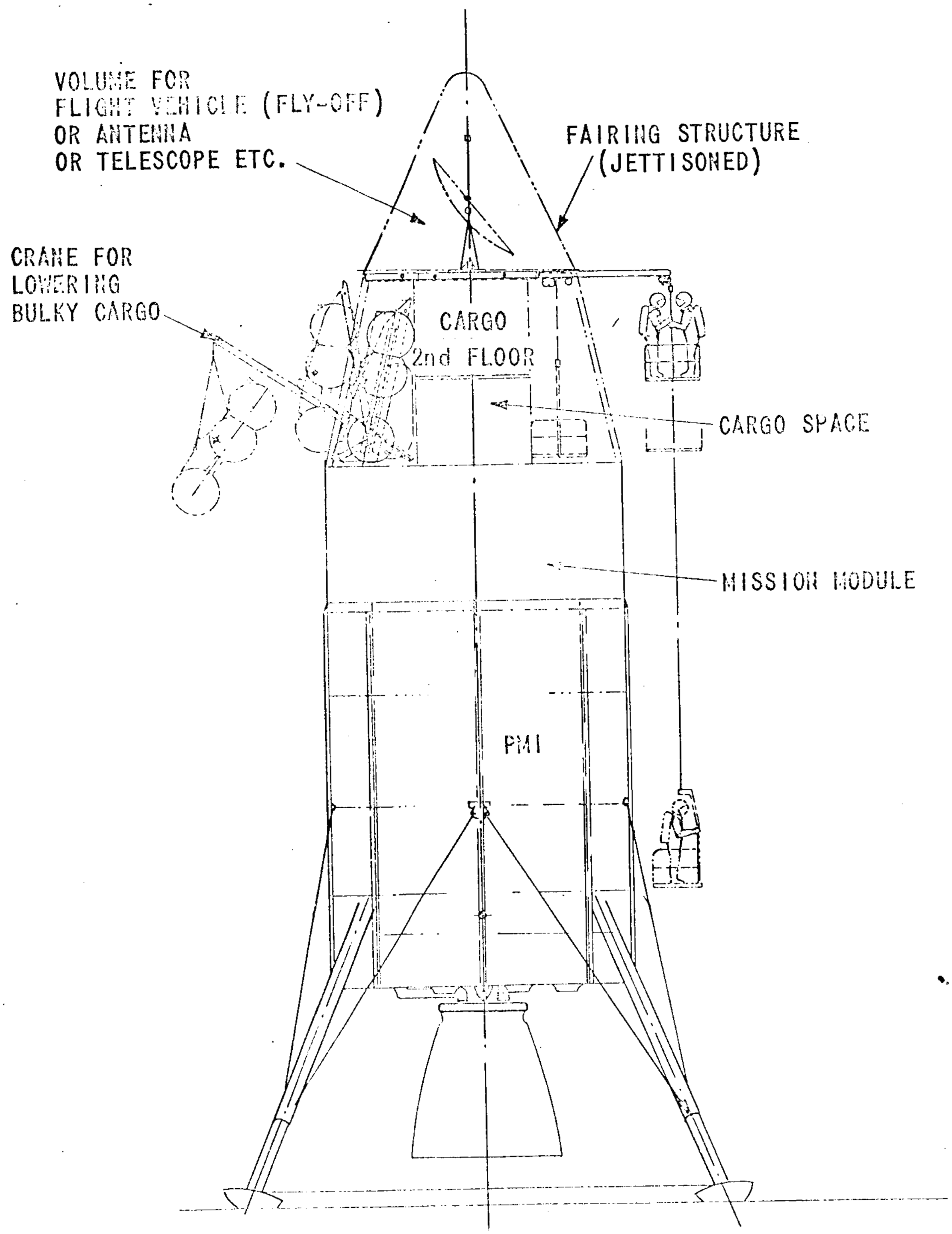
LUNAR BASE

The common hardware units can establish a long term lunar base suitable for major exploration missions. A 1-2 year base for several men and a large payload is feasible. Timing of such a base is dependent on decisions regarding mission objective emphasis and could occur anytime after hardware availability.

Base size is set by the lunar landed weight limit of the PM-I which is 45,000 lb. with a standard Saturn V (34). This base would comprise a 3 man module and more than 10,000 lb. of payload. This capability greatly exceeds that of limited stay time exploration missions utilizing shelters delivered to the surface by the Lunar module descent stage (40, 41, 42).

The landed lunar base configuration and the landing stage are depicted. Another Saturn V and lunar landing stage lands crew, earth entry module, and a lunar ascent/earth return propulsion module.

A typical cargo is shown on top of the mission module. Transfer of "suitcase size" cargo and crewmen to the lunar surface is accomplished via a small elevator. Large bulky objects, such as a roving or flying vehicle, could be lowered to the lunar surface with a crane and cable.


VOLUME FOR
FLIGHT VEHICLE (FLY-OFF)
OR ANTENNA
OR TELESCOPE ETC.
FAIRING STRUCTURE
(JETTISONED)
CRANE FOR
LOWERING
BULKY CARGO
CARGO
2nd FLOOR
CARGO SPACE
MISSION MODULE
PMI

PLANETARY MISSIONS

The extremely large weight and cost expenditures generally associated with planetary exploration demand a thorough reexamination of ground rules. The result of over-conservative design and mission mode planning can ultimately trigger added complexity and cost because of inherent multiplying effects. In this context, an approach to planetary exploration is proposed in which traditional ground rules are revised.

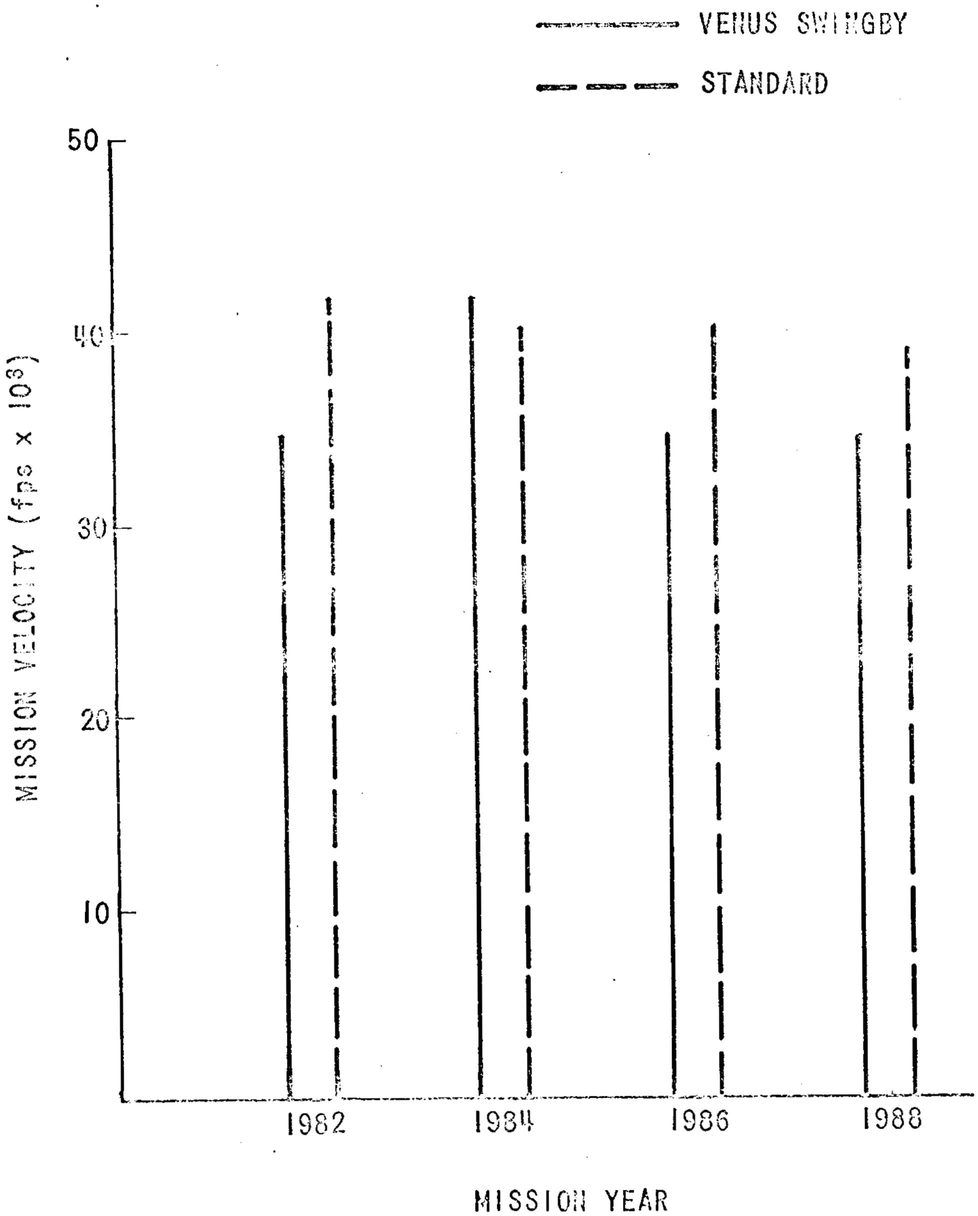
Weight and cost reductions are achieved by incorporation of all or some of the mission elements shown opposite.

MISSION ELEMENTS FOR MINIMUM SPACECRAFT WEIGHT

- MINIMUM CREW SIZE
- ELLIPTICAL PARKING ORBITS FOR EARTH ORBIT ASSEMBLY AND PLANETARY CAPTURE
- DIRECT ENTRY OF MANNED AND UNMANNED PROBES INTO PLANETARY ATMOSPHERES
- VENUS SWINGBY MODE FOR MARS STOPOVER MISSIONS
- MISSION YEARS SELECTED FOR LOW TOTAL ΔV

Venus Swingby Mode

Venus swingby modes (43, 44, 45) afford substantial reductions in characteristic velocities for manned Mars stopover missions compared to standard opposition class missions. In general, substantial reductions in mass in earth orbit result, and earth entry speeds are usually below those of other mode requirements. Total trip times (approximately 500 to 600 days) necessary to realize these benefits are only about 20% (or less) greater than standard trip times. Moreover, mission times are greatly shortened by comparison to conjunction class missions which are as much as 1,100 days, including approximately 300 days spent in the planetary capture orbit. Because of the unattractive mission duration and planetary staytime characteristics, conjunction class missions are discarded for present purposes despite the very low ΔV requirements associated with such missions. However, it is recognized that conjunction class missions could be very attractive if extended manned exploration of the Martian surface becomes an objective.



MARS 30 DAY STOPOVER MISSION MODE COMPARISON

Elliptical Earth Parking Orbit

Elliptic parking orbits for rendezvous and assembly provide substantial performance improvements over circular orbit rendezvous by allowing utilization of the full performance capability of the launch vehicle for a fixed payload. Elliptical orbits make small PM-I class planetary injection stages competitive with larger stages (which are optinal for low orbit) thus enabling commonality to be achieved with other mission classes (31). In addition to providing launch savings, simplification of orbit operations result (46). Highly eccentric orbits tuned to one or two day periods allow almost continuous communications and tracking, similar to synchronous orbit (47). Rendezvous sensitivities on the extended outbound leg may be reduced compared to circular orbit. Moreover out-of-orbit launch window penalties are significantly eased because of greatly reduced perturbations of the orbit due to the earth's oblateness as compared with low altitude circular orbits. Effects of solar and lunar gravity on lifetimes of highly elliptical orbit do not appear to be prohibitive (48).

Van Allen radiation shielding requirements in elliptical orbit are not substantially increased from low earth orbit because of the relatively quick passage through the radiation belts (49). Dose rate for a two day ellipse is less than 2 rads/day with no shielding other than spacecraft structure.

ELLIPTIC ORBIT RENDEZVOUS RATIONALE

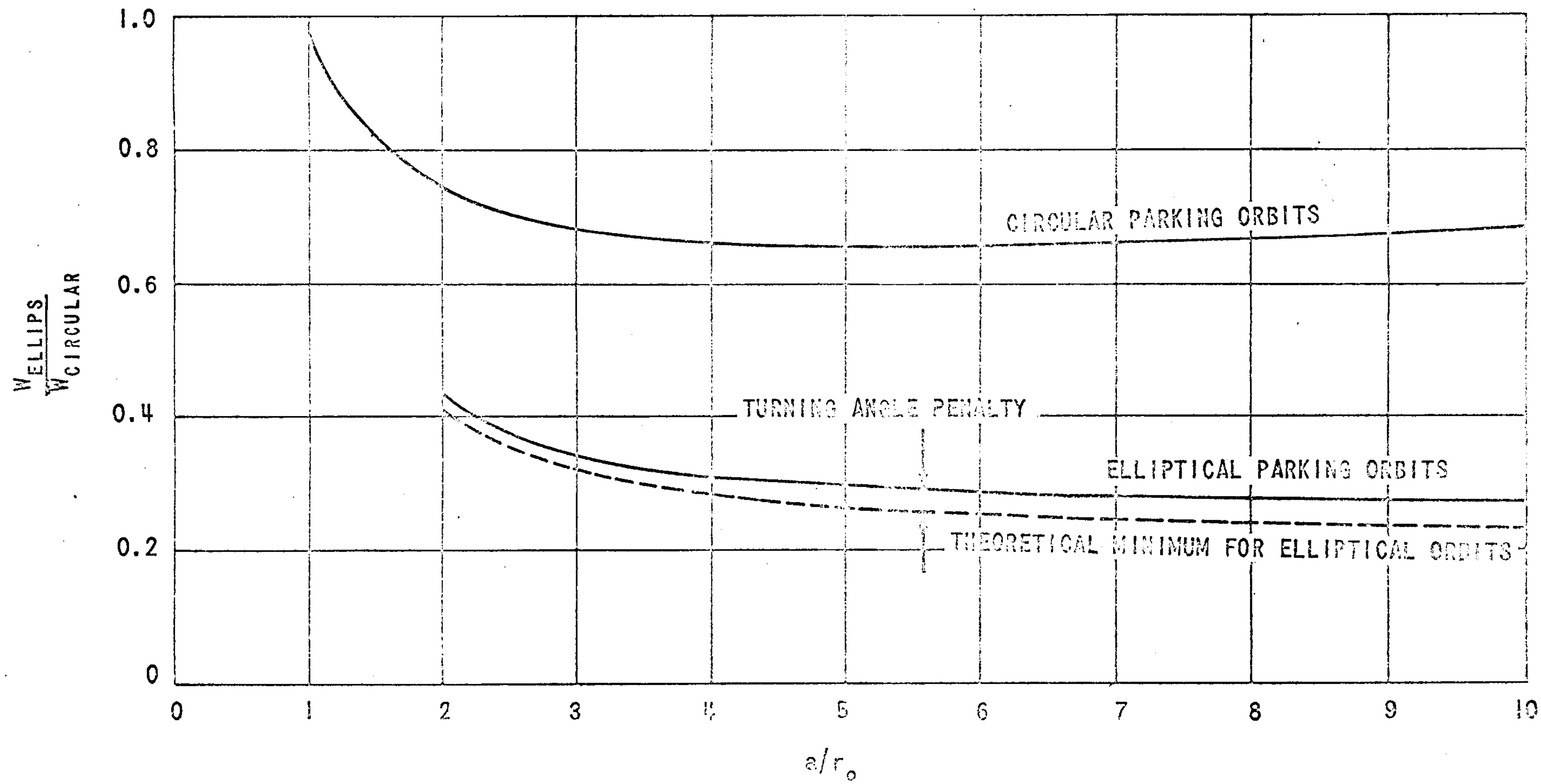
- ALLOWS UTILIZATION OF FULL PERFORMANCE CAPABILITY OF LAUNCH VEHICLES
- ALLOWS PM-1 COMMONALITY WITH LUNAR AND UNMANNED PROBE MISSIONS
- SIMPLIFICATION OF ORBITAL OPERATIONS INCLUDE:
  - ALMOST CONTINUOUS COMMUNICATIONS AND TRACKING
  - REDUCED LAUNCH WINDOW PENALTIES
- LOW VAN ALLEN RADIATION DOSAGE LEVEL IS MAINTAINED BECAUSE OF QUICK PASSAGE THROUGH BELTS

PLANETARY MISSIONS/MISSION ELEMENTS

Elliptical Capture Orbits

Elliptic capture results in substantial savings in arrival and departure velocity changes compared to circular orbit capture (50). This is demonstrated in the case of the Venus Orbiter mission where savings on the order of 50% weight reduction by comparison to circular orbit result. The savings are less for Mars percentage-wise but, in view of higher weight sensitivity for Mars capture, are equally significant.

In general, elliptical orbits markedly reduce propulsion requirements only if near theoretical minimum ΔV's (i.e., arrival and departure at periapsis) are achievable. But there are turning angle penalties for alignment of arrival and departure asymptotes. These penalties are minimized by off periapse maneuvers which are considerably less expensive than the circulatization/decircularization maneuvers. Three Mars stopover missions via Venus swingby were considered (51). In two cases the turning angle penalty is negligible ('86 and '88) and in the other case ('82) the ΔV penalty is less than 1,000 fps.



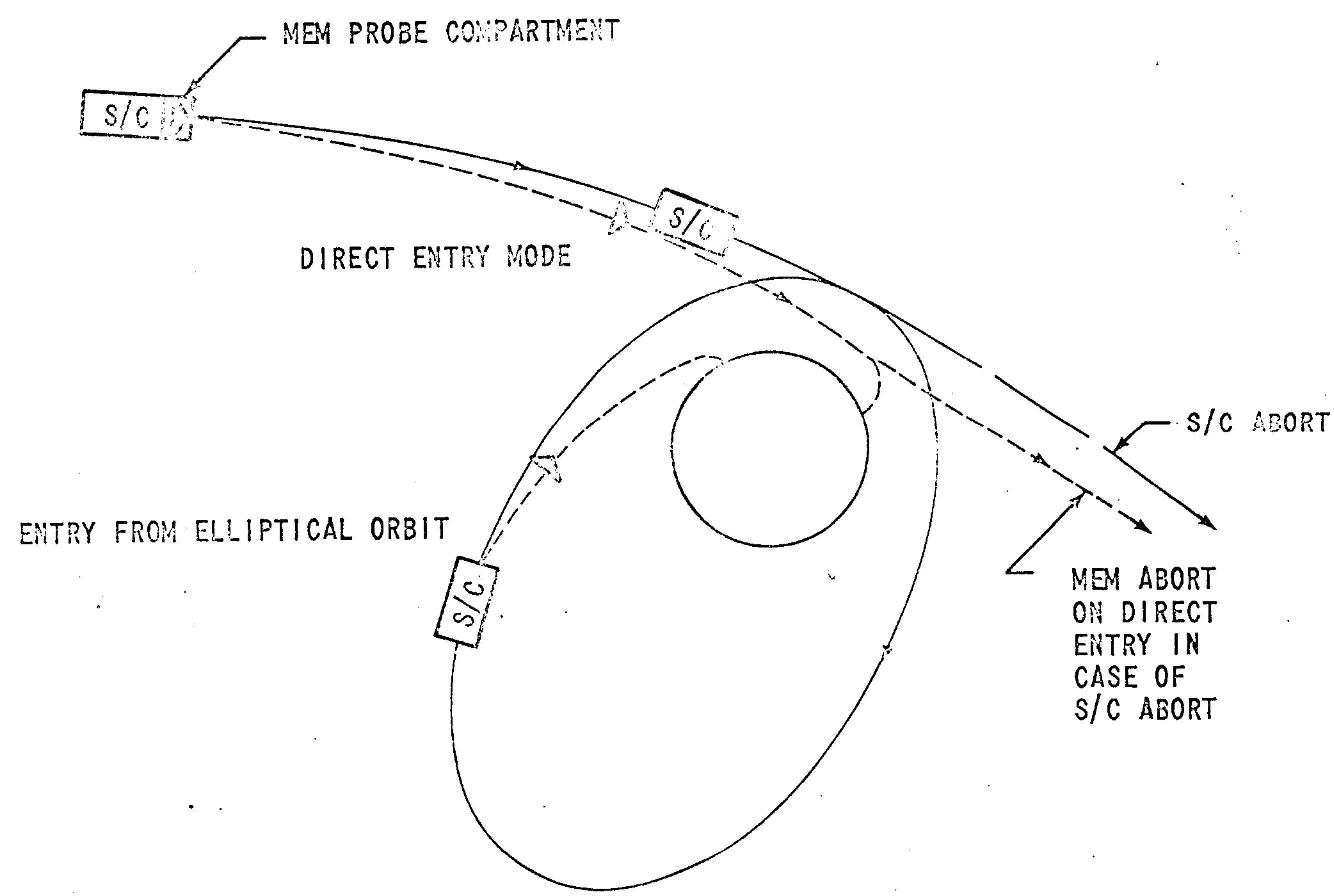
MINIMUM WEIGHT REQUIRED IN EARTH ORBIT, 1970 VENUS ORBITER MISSION

Mission Elements-MEM Direct Entry

The MEM vehicle complement (including hangar) is a considerable portion of the payload out of earth orbit, weighing approximately 50% to 80% of the manned spacecraft. A substantial penalty in added retropropellant results if the MEM is propulsively braked into planetary capture orbit. Direct entry can save as much as a SV launch by eliminating the retrobraking propellant for the MEM capture maneuver.

The deceleration limit during direct entry can be held to within 10 g's for a 10 nm. entry corridor (52), believed reasonable for the related time period. The problem is essentially a guidance one, i.e., being able to achieve the assumed 10 nm corridor tolerance. Preliminary estimates based on current capability suggest this problem will be favorably resolved upon subsequent analysis. It is not believed that the guidance problem challenges feasibility.

Abort of the manned, direct entry mission by utilization of landing stage propulsion is possible at any time up to 100 seconds before nominal entry (52). All the ascent propellant is available for rendezvous with the parent module in orbit, or chase along the heliocentric trajectory if the parent module itself aborts during capture maneuvers made prior to MEM entry. Abort after atmospheric entry is possible during 50% of the entry time.



DIRECT vs. ELLIPTICAL ORBIT MEM ENTRY MODE

PLANETARY MISSIONS/MINIMUM SCALE

MARS SURFACE EXPLORATION

Mars landing and flyby mission payloads are considered in somewhat greater detail than previous mission categories since new operational and design concepts are introduced.

As noted, Mars lander payload is a substantial fraction of total spacecraft weight, and represents a significant portion of total program development cost. A plan is proposed to accomplish principal objectives of 1) Mars surface sample retrieval and 2) manned Mars landing, at minimum overall cost. A key element of this plan is commonality of unmanned Mars surface sample return (MSSR) and manned Mars Excursion Module (MEM) probes if these two major objectives are carried out on separate missions (53).

TACTICS FOR INITIAL MANNED MARS MISSIONS

MARS SURFACE SAMPLE RETURN ON PRECURSOR FLYBY AND ORBITER MISSIONS TO BE ACHIEVED WITH BASICALLY SAME SYSTEM USED FOR MANNED LANDING.

PLANETARY MISSIONS/MINIMUM SCALE

MANNED MARS SURFACE EXPLORATION

Landing Mission Description

The interplay between the various operation and vehicle elements is best understood by a general description of a complete manned landing mission profile. Selected details are subsequently highlighted. Consistent with a minimal mission policy a two man landing mission is presumed.

The MEM vehicle arrives in the vicinity of Mars with a parent spacecraft which establishes a highly elliptical (24 to 28 hrs) capture orbit. (This orbit is non-optimum for MEM surface to orbit return but, as previously noted, is desirable to minimize parent module braking and return injection velocities.) The vehicle separates from the parent ship, and descends to the surface only after the parent spacecraft capture maneuver is achieved.

A 2 week staytime is provided during which time surface reconnaissance and experiments are performed. The astronauts return in the ascent stage and rendezvous with the parent module in elliptical orbit.

Abort capability is provided prior to entry, for a period of time shortly before touchdown, and from the surface in the event of surface shelter failure. Surface or abort launch are achieved via preprogrammed trajectories to low circular orbit. A single orbit coast (or less) is allowed for positioning and orbit determination from the parent module. Transfer is achieved so that the MEM is slightly ahead of the parent spacecraft. Initial separation is not more than several 10's of miles, and closes steadily. The MEM ascent stage is guided by radio command from the parent module during the final phases of the rendezvous sequence. At return rendezvous, the astronauts either fly the vehicle into a prepared docking area or leave the spacecraft and maneuver to the main module by EVA. A nominal mission then requires the astronauts to live in the ascent stage for perhaps 6 hours before landing and 1 or 2 hours after ascent. Upon landing the shelter is activated and the astronauts transfer to the shelter for the remainder of staytime duration.

MISSION CHARACTERISTICS

- TWO MAN MISSION
- ELLIPTICAL CAPTURE ORBIT
- TWO WEEK STAYTIME

PLANETARY MISSIONS/MINIMUM SCALE

MANNED MARS SURFACE EXPLORATION

Mars Lander Vehicle Description

The MEM is comprised of a cone or Apollo shaped entry shell which contains heat shield, retropropulsion, landing gear; the MEM ascent stage; and a laboratory/shelter for surface operations. The MEM ascent stage houses descent command system control interfaces, the ascent capsule, and return propulsion stages. Abort considerations necessitate that the astronauts ride in the MEM ascent stage during entry to allow rapid escape.

The achievement of minimum MEM gross weight is contingent upon minimizing ascent (Mars surface to orbit) vehicle weight. To accomplish this the ascent capsule is designed solely to provide transportation to and from the surface of Mars to the parent spacecraft in parking orbit (54). No experiments are performed en route and communications and telemetry are minimal. There are no operations that require man's functional mobility, so compact packaging of the ascent capsule can be achieved.

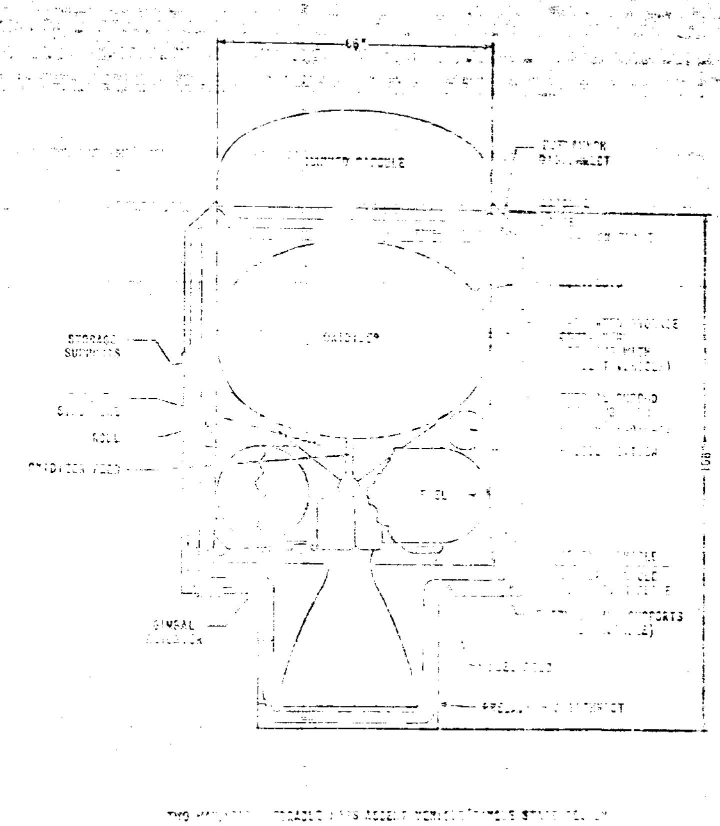
When possible subsystems are located in the descent stage. For example, the relatively heavy entry landing system (i.e., computers, guidance, and communications subsystems) are packaged in the descent stage and connected to the ascent/command capsule by umbilicles capable of being broken immediately in case of launch for abort.

In the ascent stage capsule life support and environment control are to be provided entirely by the suit loop. (Precursor lunar surface EVA experience should preclude the necessity for independent cabin EC/LSS backup.) Landing maneuvering is accomplished with remote sensing by landing TV and video display. This allows more efficient placement of the ascent capsule and minimum packaging design.

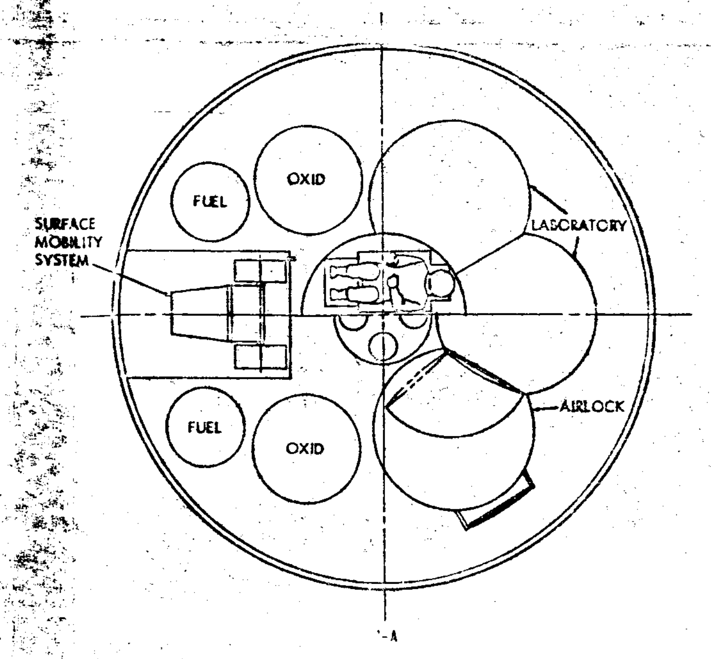
During ascent continuous communications is provided down range by the parent spacecraft to which the MEM can respond via omni antenna by a minimal up link communications system utilizing equipment in the back pack (55). To reduce weight in the ascent capsule, radio command guidance (from the parent spacecraft) is employed for ascent return rendezvous (56).

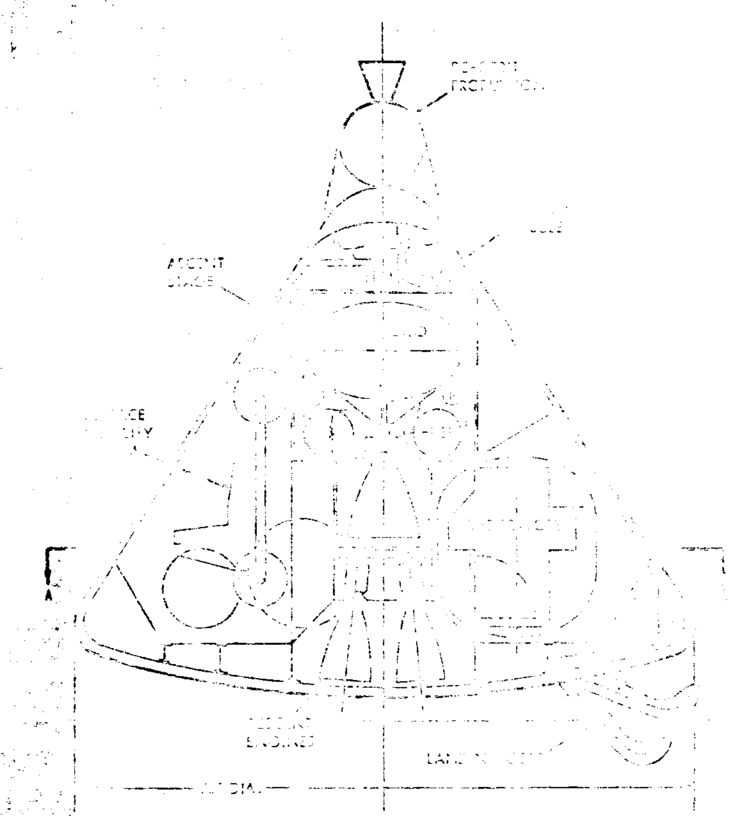
Upon consideration of these and other subsystem requirements (i.e., power, RCS, etc.) it has been estimated that the weight of a 2 man manned ascent capsule is of the order of 1300 lbs (54).

THE ASCENT PROPULSION STAGE (57) IS SIZED FOR A TOTAL VELOCITY OF 19,000 fps; ENOUGH TO LAUNCH FROM THE SURFACE AND RENDEZVOUS WITH THE MANNED SPACECRAFT IN A ONE DAY PARKING ELLIPSE (23 1/2 HOURS) WITH ADEQUATE COURSE CORRECTION VELOCITY (58). SPACE STORABLE FLOX/METHANE PROPELLANTS ARE UTILIZED. (NUMEROUS FACTORS WERE INVOLVED IN THES SELECTION i.e., STORABLE PACKAGING, LAUNCH PAD HANDLING, ETC. AND ARE DISCUSSED IN REFERENCE 57.) IT IS ESTIMATED THAT THE CROSS WEIGHT OF A TWO MAN SINGLE STAGE FLOX/METHANE MEM ASCENT STAGE IS APPROXIMATELY 7500 LBS.



FOLDOUT FRAME 1





FOLDOUT FRAME 2

2 MAN MEM DESCRIPTION

- A MINIMUM TWO MAN PAYLOAD CAPSULE DESIGNED SOLELY TO PROVIDE TRANSPORTATION TO THE SURFACE OF MARS CAN BE DESIGNED FOR AN ESTIMATED WEIGHT OF 1300 LBS (INCLUDING THE MANNED SYSTEMS)

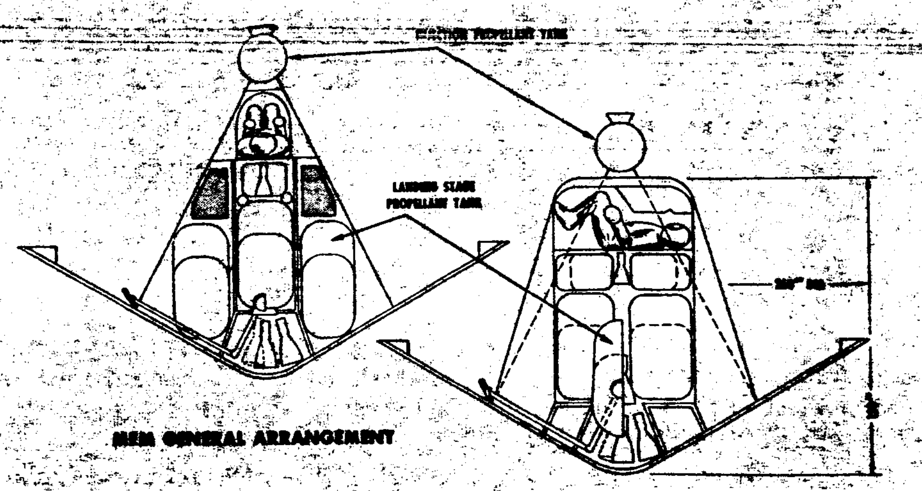
- ASCENT VEHICLE DESIGN DESCRIPTION
  - SUIT LOOP ENVIRONMENTAL CONTROL LIFE SUPPORT SYSTEM (NO CABIN BACKUP)
  - CONTINUOUS DOWNRANGE COMMUNICATIONS BUT LIMITED UPRANGE VOICE LINK
  - RAPID COMMAND GUIDANCE FOR RETURN RENDEZVOUS

PLANETARY MISSIONS/MINIMUM SCALE

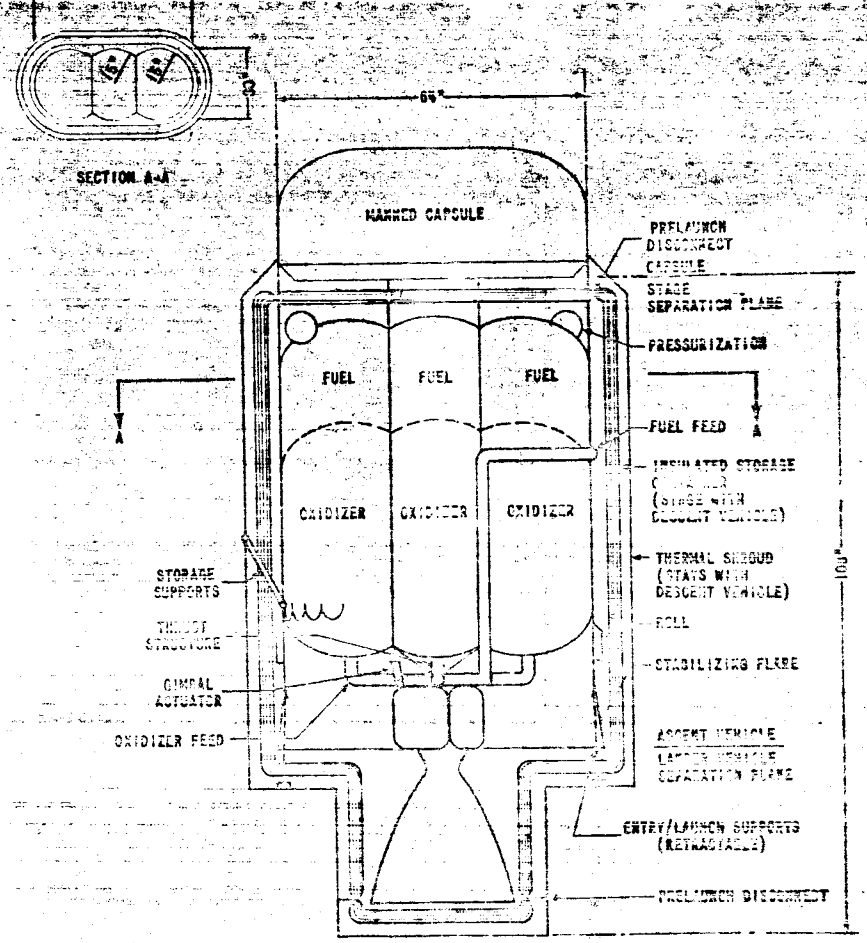
MANNED MARS SURFACE EXPLORATION

Alternate Approach Employing Surface Rendezvous

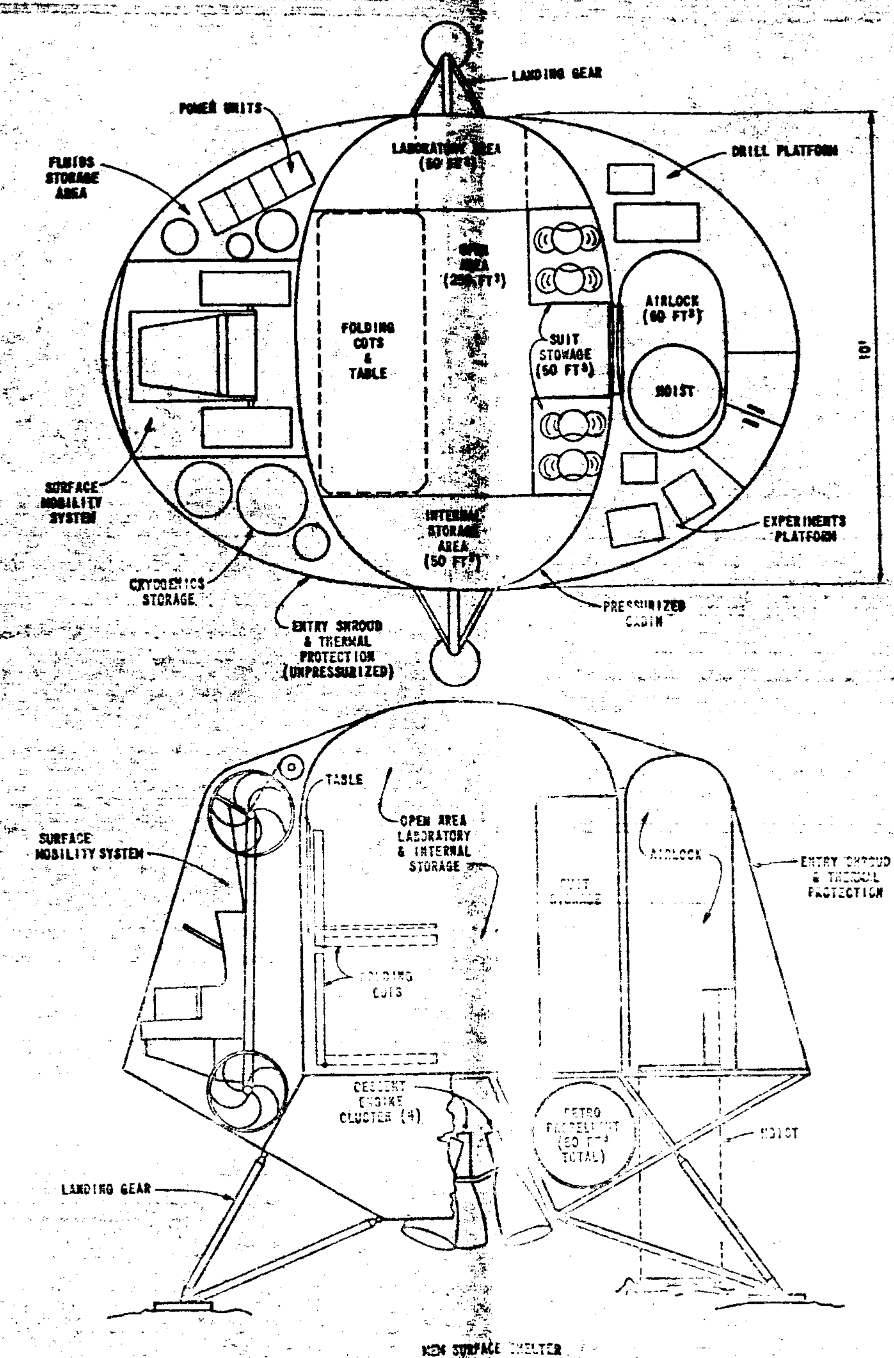
Major surface missions, involving several crewmen and arbitrary surface staytimes could be accomplished utilizing small separate spacecraft which rendezvous on the Martian surface, in lieu of a larger combined system. (Separate vehicles would carry either a single crewman or an unmanned surface shelter.) Cost savings resulting from size scaledown advantages, and mission scale flexibility are inherently achieved. (For example, an early Mars landing mission could employ a single crewman for several hours staytime.)



MEM GENERAL ARRANGEMENT



ONE MAN/SPACE STORABLE MARS ASCENT VEHICLE/SINGLE STAGE DESIGN



MEM SURFACE SHELTER

TYPICAL MISSIONS

VEHICLE DESCRIPTION

THE ONE MAN CAPSULE IS APPROXIMATELY A HALF CYLINDER SIZED TO ACCOMMODATE A SINGLE ASTRONAUT. TO MINIMIZE DRAG DURING ASCENT (SO) PROPELLANT IS PACKAGED COMPLETELY WITHIN THE PAYLOAD SHADOW. THE WEIGHT OF THE ONE MAN ASCENT STAGE IS APPROXIMATELY 4,300 LBS (S7).

THE SHELTER VEHICLE HAS A PAYLOAD ALLOTMENT OF APPROXIMATELY 9,000 LBS WHICH IS SUFFICIENT FOR EXPERIMENTS AND EXPENDABLES FOR A 2 MEN-2 WEEKS STAYTIME. THIS IS CONSISTENT WITH THE ESTIMATED WEIGHT OF A MANNED ASCENT VEHICLE (CO) SO THAT COMMON ENTRY SYSTEMS (i.e., HEAT SHIELD, LANDING GEAR, AND RETROPROPULSION) CAN BE EMPLOYED.

FOLDOUT FRAME A

FOLDOUT FRAME B

PLANETARY MISSIONS/MINIMUM SCALE

MANNED MARS SURFACE EXPLORATION

Alternate Approach - MEM Gross Weight Breakdown

Gross weight of MEM for direct entry is estimated to be three times landed payload (52). Gross weight of the combined two-man vehicle is about 35,000 lbs. (Note: This is significantly lighter than previous concepts (61,62) because of minimal weight ascent capsule design.) The small vehicle with 4,300 lbs landed payload weighs about 13,000 lbs. Gross weight for a three vehicle mission is therefore approximately 38,000 lbs.

Probe hangar weight is estimated to be approximately 13,000 to 15,000 lbs so that gross weight charged to MEM is 48,000 to 53,000 lbs.

FEATURES OF SURFACE RENDEZVOUS EXPLORATION MODE

- SMALL VHEICLES WITH ASSOCIATED COST SCALEDOWN

- FLEXIBLE MISSION PLANNING

PLANETARY MISSIONS/MINIMUM SCALE

MANNED MARS SURFACE EXPLORATION

Alternate Approach Employing Surface Rendezvous - Mission Profile

The mission profile for a two man mission is described for comparison with the baseline single vehicle approach. Here two single man vehicles and a separate shelter vehicles are employed.

The three vehicles descend simultaneously to the surface. Aerodynamic characteristics of the three vehicles are identical and all follow the same preprogrammed entry profile. The entry/landing mode can have a wide individual landing footprint but the relative landing dispersions, the critical element for surface rendezvous, are small (52). A ΔV contingency is allocated for approximately 1 minute hover and translation (in addition to retrobraking contingency) as further assurance against excessive landing separation. (Propulsive range make up employing horizontal translation is on the order of 10% of the vehicle weight per mile). Alternately the shelter craft can enter the Martian atmosphere several minutes ahead of the manned vehicles so that the manned craft can target to the shelter landing site.

Should the shelter stage become inoperative, or separation distance be found excessive, the astronaut can return via the ascent vehicle after a curtailed staytime of 1 day, or less, with surface samples and limited data obtained at the landing site. Hence, the mission can be rated a partial success even without use of the shelter since abort is continuously available via a separate vehicle.

MEM WEIGHT BREAKDOWN

	ONE MAN VEHICLE	TWO MAN VEHICLE AND SHELTER
MANNED CAPSULE	700	1300
ASCENT PROPULSION STAGE	3500	6200
SURFACE SHELTER	---	4000
DESCENT STAGE	8400	23000
VEHICLE WEIGHT	12,600	34,000
NUMBER OF VEHICLES	3	1
TWO MAN MISSION WEIGHT	~ 38,000	~ 35,000
STORAGE HANGAR MODULE WEIGHT	15,000	13,000
WEIGHT IN EARTH ORBIT	53,000	48,000

PLANETARY MISSIONS/MINIMUM SCALE

MANNED MARS SURFACE EXPLORATION

MSSR/MEM Commonality

A prime goal of precursor Mars orbiter and flyby missions (prior to manned landing) is to search for and identify life on Mars via surface sample return. A Mars Surface Sample Retriever (MSSR) probe (63) in association with a manned spacecraft, could accomplish surface sample recovery from a selected site, and enable analysis onboard the spacecraft immediately upon recovery. To minimize costs, the system that is used for this initial goal can be a developmental and test version of the manned landing system. The precursor unmanned mission could then support manned landing goal, not only in discovering possible life forms, but also in developing the necessary hardware.

Developing a manned lander will permit considerable payload for unmanned sample return on much higher ΔV flyby and orbiter missions (53, 64). This enhances the chance of successful unmanned retrieval system design. The payload "pad" varies with the particular mission selected and is more than adequate for unmanned sample retrieval. Moreover, this "pad" permits flexibility of mission selection. Advantages of such an approach are single development of the descent lander stage and ascent propulsion stages, and qualification of a single vehicle. Improved reliability for the manned mission is also achieved through increased usage during "all up" MSSR operations on unmanned missions.

A further advantage of this approach is that flight experience gained from unmanned missions (65) would allow precise calibration of performance characteristics and targeting accuracy in the real Martian environment to enhance operations of later manned flights. (Gemini experience provides instructive example. Here reentry targeting accuracy was continuously improved with flight experience. Likewise the Mars entry vehicle guidance and control would be uprated to improve atmospheric entry and surface site targeting accuracy.)

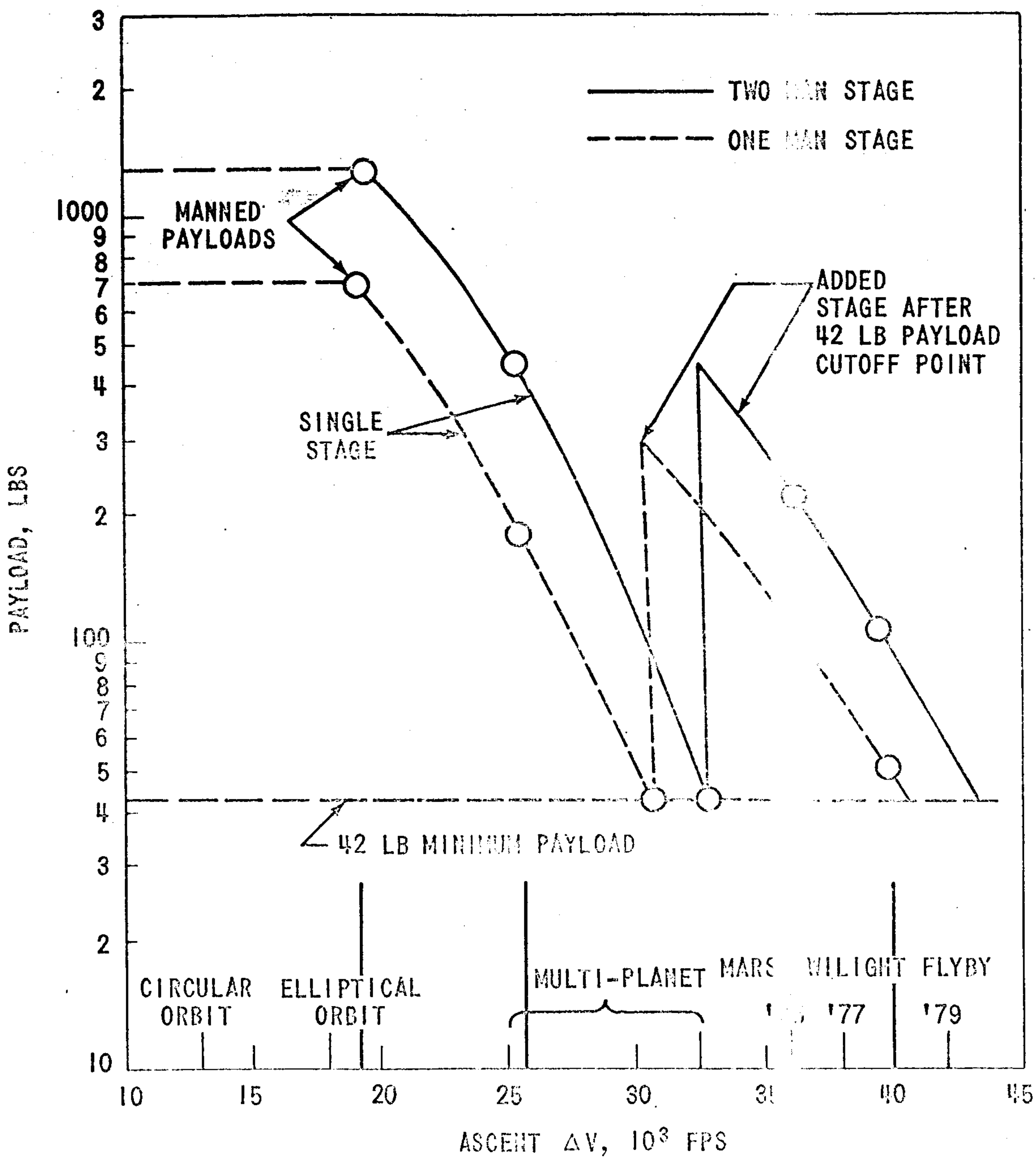
MSSR/MEM COMMONALITY

- SINGLE DEVELOPMENT OF DESCENT LANDER STAGE AND ASCENT PROPULSION
- SINGLE QUALIFICATION
- RELIABILITY THROUGH EXTENDED USAGE

MANNED MARS SURFACE EXPLORATION

MSSR/MEM Commonality

The figure opposite shows velocities achieved by unmodified one man and two man ascent stages, operating in an unmanned mode for a fixed MSSR mission with the 42 lb payload derived in Reference 63. All vehicles achieve velocities greater than 31,000 fps; more than sufficient for low energy classes of dual and triple planet flyby missions (57). Velocities for these missions would more typically be on the order of 26,000 fps in which case payloads of 185 lb and 430 lb could be achieved by one and two man vehicles, respectively. Higher energy (or payload) missions such as the Mars "twilight" flyby (single-planet) in the 36,000 fps to 40,000 fps class require an added stage. The stepped curve shows MEM launch capability with the added stage plus payload gross weight equal to the weight of the manned capsule. Payloads are 77 and 212 lbs for 36,000 fps velocity and 27 and 102 lbs for 40,000 fps velocity respectively for the two types of vehicles.



PAYLOAD VS. ASCENT VELOCITY

PLANETARY MISSION/POTENTIAL MANNED MARS

LANDING MISSION WEIGHT REDUCTIONS

To illustrate the extent of savings attributed to each mission element, percent reductions of initial weight in earth orbit (IWIEO) are presented for a sample planetary mission (1982 manned Mars landing) (66).

The basic mission model assumes departure from a low-altitude circular orbit at Earth,* propulsive braking at Mars, descent of an excursion module to the surface of Mars, eventual rendezvous in orbit with the main spacecraft, propulsive departure from Mars (after jettisoning the excursion module) and entry into the Earth's atmosphere at a speed not exceeding 50,000 fps.

Effects of the following mission profile variations are compared:

1. Venus swingby versus direct Mars-Earth return leg.
2. Highly eccentric (~1-2 1/2 day period) parking orbit versus low-circular orbit at Mars.
3. Direct entry of the MEM (Mars Excursion Module) from hyperbolic approach; i.e., separation from the mission module before entering the Mars capture orbit, versus entry from orbit.
4. Crew of four versus crew of six.
5. MEM crew of two versus crew of three.

Mission ΔV's are selected to allow for approximately twenty day launch windows at both earth and Mars.

The "conventional" mission (67) involving circular parking orbit at Mars and crew of six men, requires about 3.5×10^6 lbs IWIEO for the direct mission or about 2.8×10^6 lbs for Venus swingby. Using elliptical orbits at Mars and direct entry for MEM, IWIEO drops to about 1.3×10^6 lbs. Incorporating crew size reductions, IWIEO with a four man crew is 830,000 lbs (within the launch capacity of three product improved SV's).

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*To establish common earth launch conditions.

Sensitivity to MEM weight is somewhat less pronounced in the 1982 mission than in general because of the low Mars arrival speed and substantially higher departure speed associated with this particular year. In many other mission opportunities the arrival and departure speeds are more evenly matched and in other cases the arrival speed is substantially higher than the departure speed. In those years sensitivity to MEM weight and the effect of MEM separation prior to orbit capture will be notably increased.

PERCENT WEIGHT REDUCTION OF INITIAL WEIGHT IN EARTH ORBIT

MISSION MODE EFFECT	IWIEO REDUCTION(%)
ELLIPTICAL ORBIT VERSUS CIRCULAR ORBIT	51
VENUS SWINGBY VERSUS OPPOSITION CLASS MODE	18
DIRECT ENTRY VERSUS ORBITAL ENTRY FOR MEM	8
TOTAL MISSION MODE REDUCTION	63
CREW SIZE EFFECTS	
SPACECRAFT WEIGHT (CREW OF 4 VERSUS CREW OF 6)	30
MEM WEIGHT (CREW OF 2 VERSUS CREW OF 3)	6
TOTAL CREW SIZE REDUCTION	36
TOTAL REDUCTIONS (MISSION MODE PLUS CREW SIZE)	76

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